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# JET PROPULSION

*Journal of the*

— AMERICAN ROCKET SOCIETY —

*Rocketry . . . . Jet Propulsion Sciences . . . . Astronautics*

STACKS

VOLUME 25

SEPTEMBER 1955

(In Two Parts)

SCIENCE AND TECHNOLOGY  
NUMBER 9, PART 1

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## Scope of JET PROPULSION

JET PROPULSION, the Journal of the American Rocket Society, is devoted to the advancement of the field of jet propulsion through the publication of original papers disclosing new knowledge and new developments. The term "jet propulsion" as used herein is understood to embrace all engines that develop thrust by rearward discharge of a jet through a nozzle or duct; and thus it includes systems utilizing atmospheric air and underwater systems, as well as rocket engines. JET PROPULSION is open to contributions, either fundamental or applied, dealing with specialized aspects of jet and rocket propulsion, such as fuels and propellants, combustion, heat transfer, high temperature materials, mechanical design analyses, flight mechanics of jet-propelled vehicles, astronautics, and so forth. JET PROPULSION endeavors, also, to keep its subscribers informed of the affairs of the Society and of outstanding events in the rocket and jet propulsion field.

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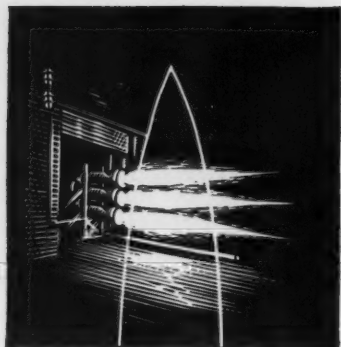
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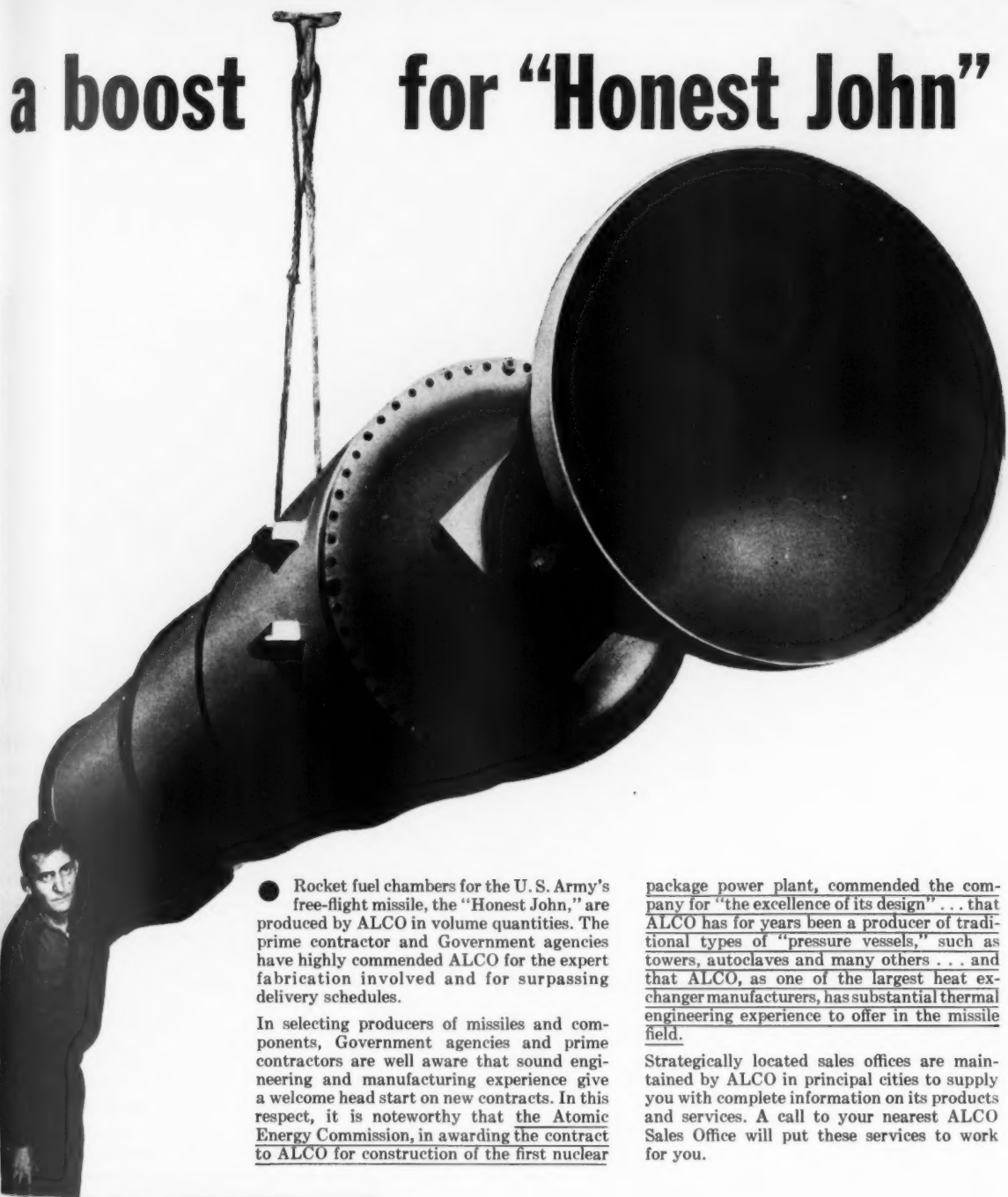
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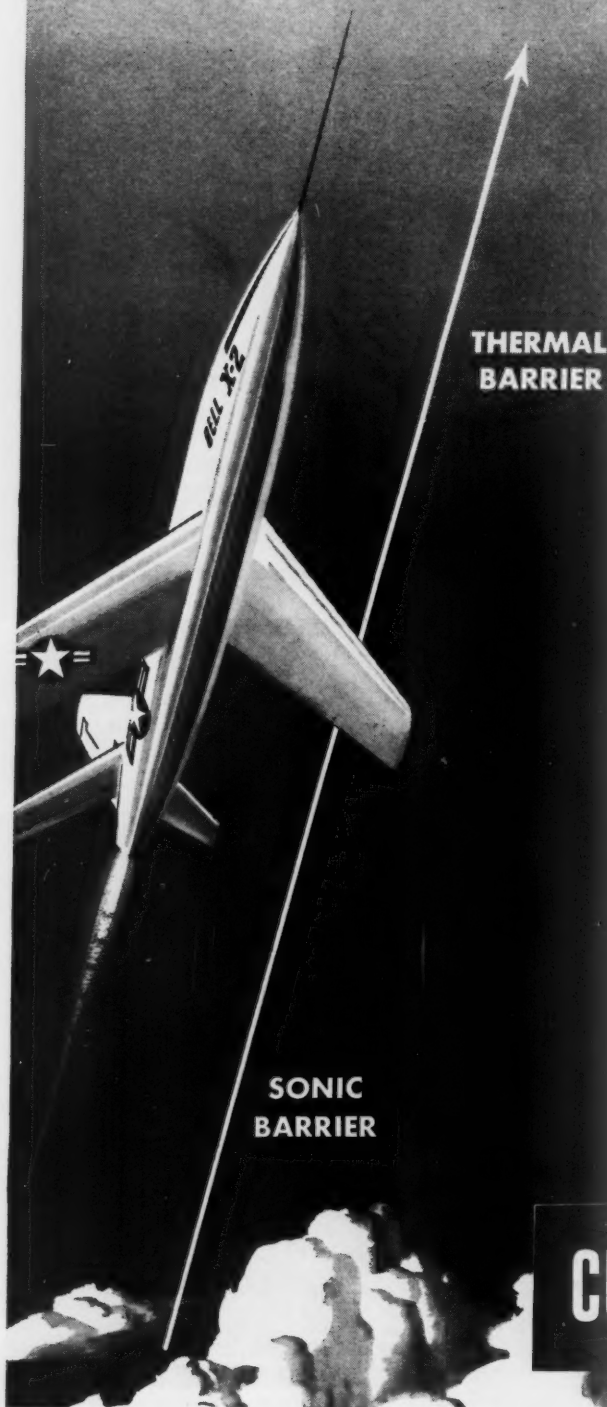
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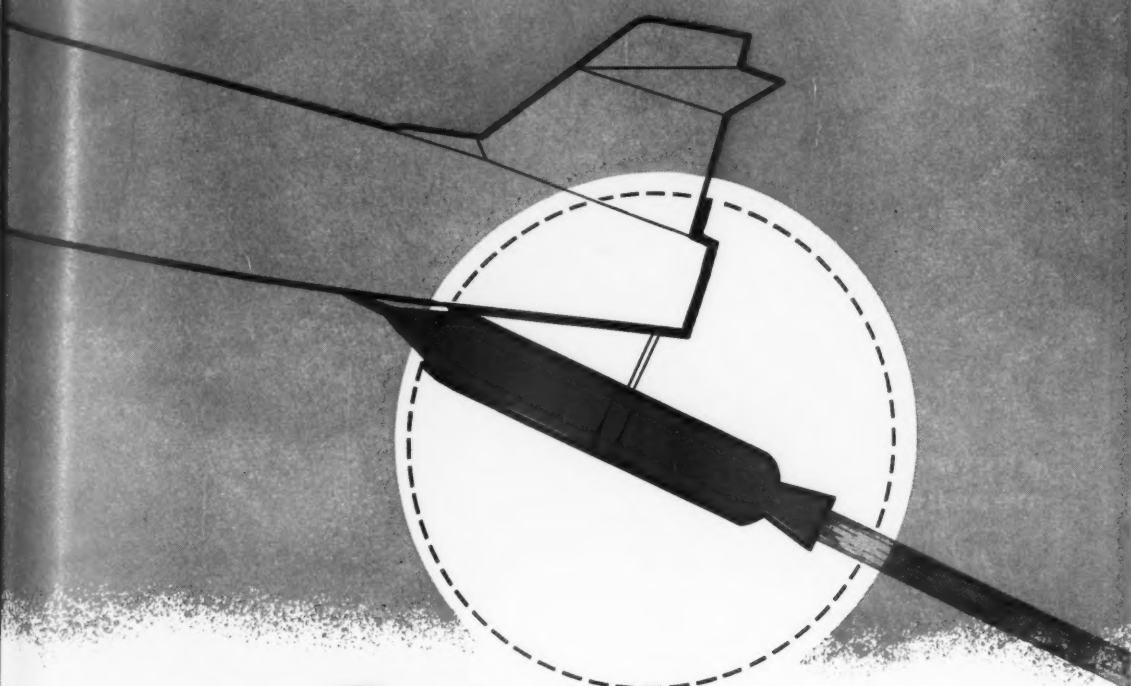
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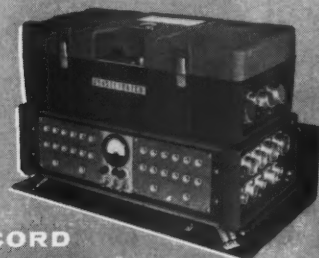
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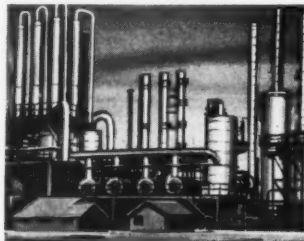
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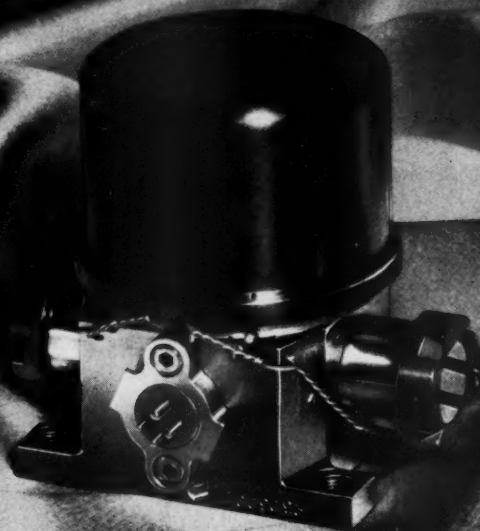
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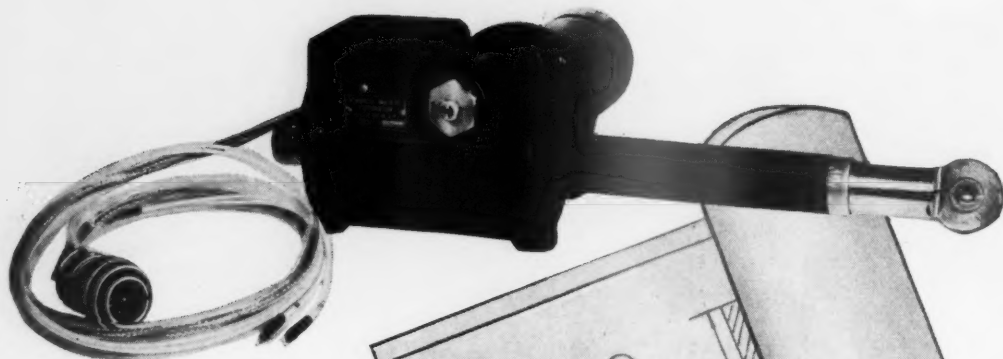
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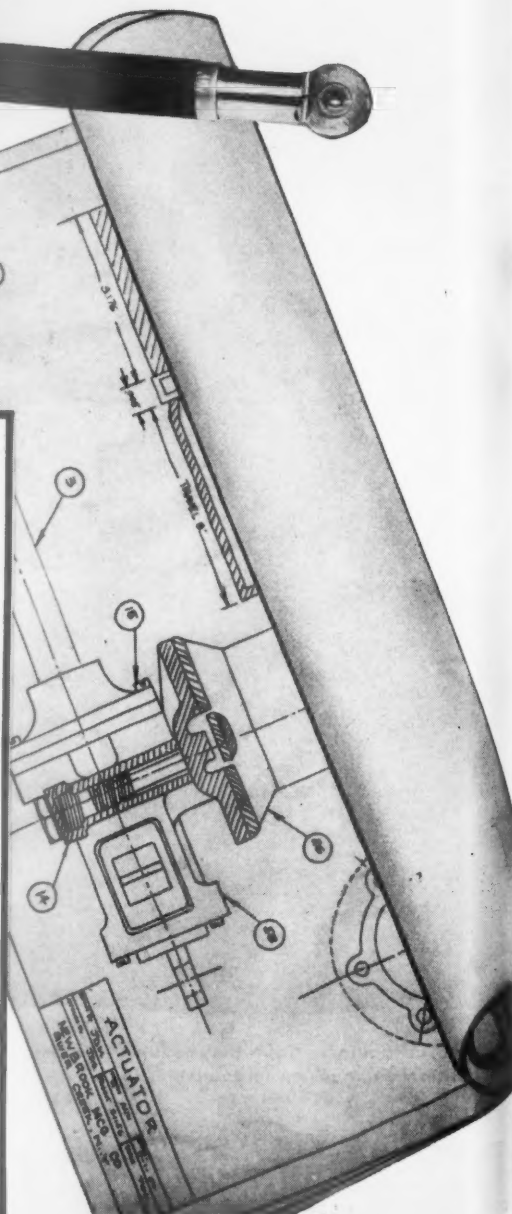
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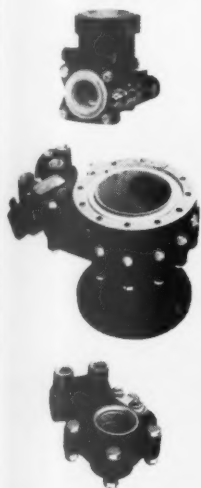
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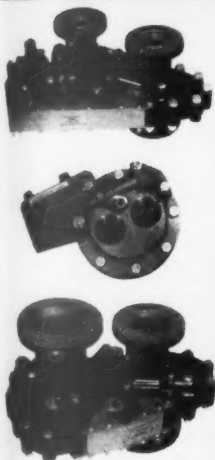




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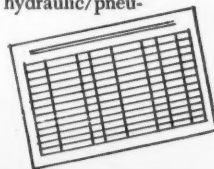
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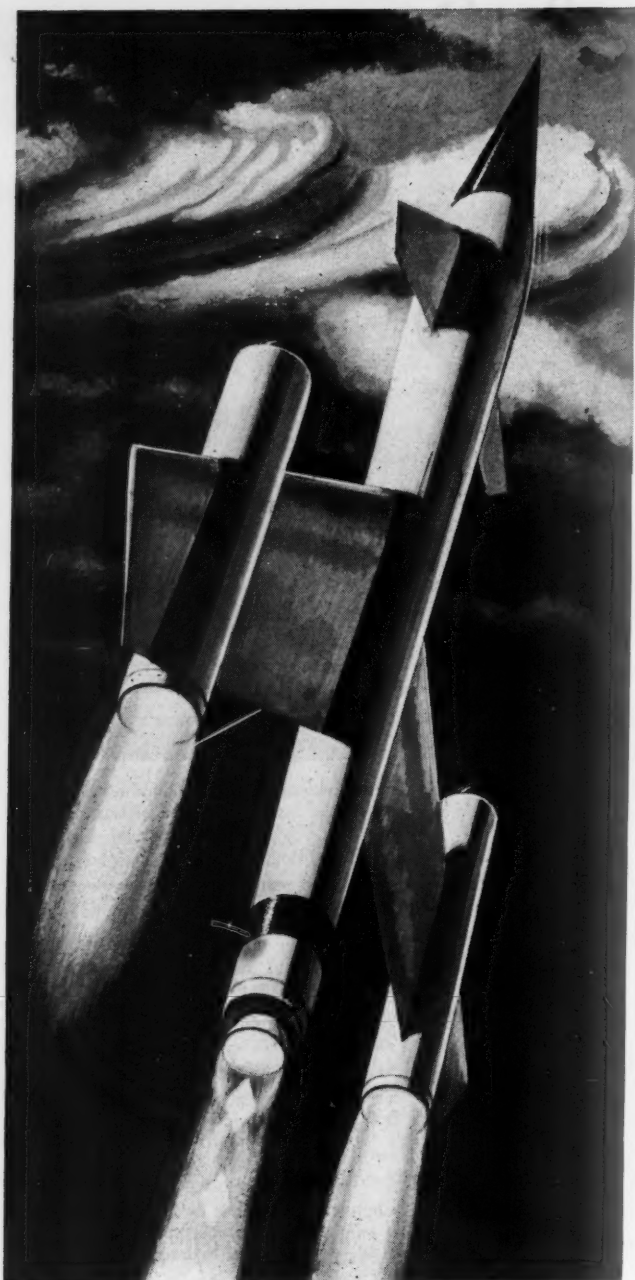
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# Rocket Powered Test Vehicles<sup>1</sup>

D. L. ARENSON,<sup>2</sup> J. H. McCLOW, Jr.,<sup>3</sup> and H. W. WIAIT<sup>4</sup>

Cook Research Laboratories, Cook Electric Co., Skokie, Ill.

In the design and development of present-day aircraft and missile equipment, it is frequently essential to obtain tests of particular components at supersonic speeds and high altitudes. For these purposes a series of rocket powered test vehicles have been developed for use on the Free Air Facility Track at Edwards Air Force Base, Calif. These vehicles are described, the basis for their design presented, and their performance is analyzed. Rocket powered missiles have also been used to study such problems as ejection seat operations and recovery parachute operations at supersonic speeds. Considerations pertinent to the design of these vehicles are described and an evaluation of their performance is presented.

## Introduction

In the design and development of present-day aircraft and missile equipment, it is frequently essential to obtain tests of particular components under conditions closely simulating those encountered during operations. For modern aircraft, this requires testing at supersonic speeds and at relatively high altitudes. Although tests can be conducted on small scale models in a wind tunnel installation, this type of testing must still be conducted under comparatively artificial conditions.

## Role of the Free Air Test Facility

Free air facility test installations have been used successfully for determining the aerodynamic characteristics and performance at transonic and supersonic speeds of such varied shapes as parachutes, bombs, flutter models, and the entire empennage of a fighter aircraft.

Basically, a free air facility installation consists of a straight precision aligned track similar to a conventional railroad.

The model to be tested is mounted on a rocket powered test vehicle usually referred to as a sled because it slides along the track on stellite-lined skids or slippers.

A track-type test installation has many advantages. Large or even full scale models can be tested; thus, dimensional scale factors need not be considered. Tests in which an operational event, such as a parachute deployment or a seat ejection occurs, can be conducted without the difficulties that would be encountered in the confined space of a wind tunnel. In addition, there are no tunnel blockage problems.

A free air facility installation is relatively inexpensive to operate, and the initial cost is much less than that of a wind

tunnel with comparable performance. Up to the present time, the scheduling difficulties on tracks have been much less severe than in wind tunnels with similar capabilities.

Track installations also have some disadvantages. The actual usable test periods are of short duration. They are limited by the length of the track and by the time during which the sled can be maintained at required test velocity. The effects of lateral and vertical vibration, acceleration, and cross winds must be considered. Although a single test is relatively inexpensive, preparations for a test may require considerable time. Also questions of validation at altitude under actual flight conditions remain unanswered.

One solution to this problem is the use of inexpensive component test missiles. Briefly then, let us consider each of these basic test methods.

First, a look at the free air facility. Tracks range in length from the 2000-ft North Track at Edwards Air Force Base to the 4.1 mile SNORT track at NOTS, Inyokern, Calif. The gage of these tracks varies from standard railroad gage to a three rail system with an over-all width of 8 ft.

Rocket motors that are used to power the test sleds range from complex liquid fuel motors to simple solid propellant units similar to the booster or JATO units used on guided missiles.

Free air facility test techniques have proved to be particularly successful for testing parachutes under operational conditions. The requirement for a satisfactory method of testing parachutes arose in connection with an Air Force program to develop parachute recovery systems for high speed missiles and target aircraft. The initial phase of this program was concerned with the development of a small, heavy-duty parachute which could be deployed at transonic speeds. For the reasons discussed above, a free air facility was chosen to provide the experimental information essential to this type of development.

The 10,000-ft track at Edwards Air Force Base was made available for use on the project, and two test vehicles were designed and built for use at this installation.



Fig. 1 Subsonic parachute test

The vehicle shown in Fig. 1 towing a Ribbon-type parachute was used in the early phases of the program. This sled was powered by solid propellant JATO units, each with a thrust of 10,000 lb for two seconds. The maximum speed of

<sup>1</sup>Presented at the ARS Ninth Annual Convention, December 1, 1954, as two separate papers: "Track Borne Test Vehicles" and "Supersonic Component Test Missile."

<sup>2</sup>A majority of the work reported in this paper was performed under Air Force Contracts AF 33(616)-448 and AF 33(038)-10653.

<sup>3</sup>Chief Engineer, Excel Industries, Inc., Chicago, Ill. Mem. ARS.

<sup>4</sup>Project Engineer, Cook Research Laboratories.

<sup>5</sup>Project Engineer, Cook Research Laboratories.

the sled was adjusted by varying the number of JATO bottles used. As many as six bottles have been used on one test run. The JATO units which were used in the early tests were obtained from a stockpile of units which had been condemned for missile or aircraft use after having been used in a series of environmental tests. On several occasions, these condemned units exploded during ignition and severely damaged the test sled (see Fig. 2). In later tests, damage to the test sled due



Fig. 2 Malfunction of rocket units

to explosions was minimized by mounting the condemned units on an independent pusher sled of very simple and inexpensive construction. When maximum performance was required, sustainer JATO's of a reliable type were mounted on the test sled.

Figs. 3 and 4 show the test sled at Station Zero during pre-test activities. The parachute to be tested is packed in the rear of the capsule above the test vehicle. The JATO units are ignited simultaneously from a blockhouse near Station Zero, and the sled accelerates to the required test speed. A knife blade is mounted on the track at the station where it has been predicted that this test speed will occur. This knife cuts a wire on the sled which in turn actuates a powder charge that deploys the parachute to be tested. This use of position sequencing permits cameras to be accurately placed along the

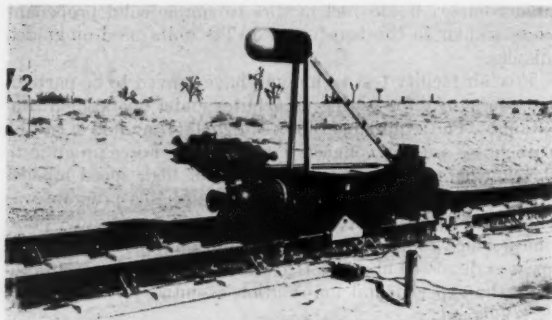


Fig. 3 Subsonic parachute test vehicle



Fig. 4 Subsonic parachute test vehicle

track at the precise location where an important action is to occur.

The capsule also contains a multichannel magnetic tape recorder developed by the Cook Research Laboratories on which the output of sensing elements measuring various pertinent aspects of parachute and vehicle performance are recorded. Parachute drag force is measured by use of a strain gage type tensiometer mounted between the parachute riser and the attachment to the sled.

Permanent magnets are located at fixed intervals along the track. Inductive pickups mounted on one of the slippers provides accurate space-time relations. Impact pressure measurements also are recorded by using a conventional pilot tube. In addition, the output of various accelerometers mounted on the sled is recorded.

A Fastax camera operating at 1000 frames per second and mounted at the rear of the test sled provides graphic information about parachute stability and performance.

The typical film strip from this camera shown in Fig. 5 portrays an asymmetric opening process of an 8-ft diameter FIST Ribbon parachute. The time interval between the frames shown is approximately 8 milliseconds.

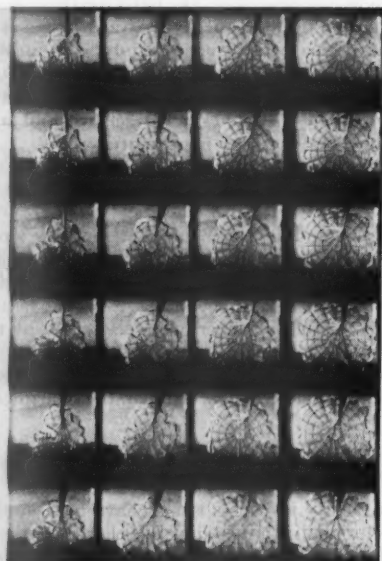


Fig. 5 Sequence photograph from sled-borne camera

Fig. 6 presents an isometric sketch of a typical slipper used on the sled. It is fabricated from steel and lined with stellite pads. A single slipper has been used for as many as fifty tests without replacement. Major wear appears to occur in dragging the sled back along the track rather than during the actual test runs at high speed.

Note that the slipper grips the rail so that upward as well

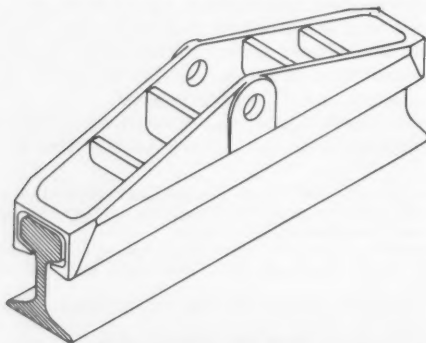


Fig. 6 Slipper configuration



as downward forces may be transmitted to the rails. Design of the track is such as to permit upward vertical forces as high as 15,000 lb on each slipper. Allowable lateral loads are 5000 lb per slipper.

The test sled is decelerated at the end of the track by means of a specially designed water brake (Fig. 7). A continuous

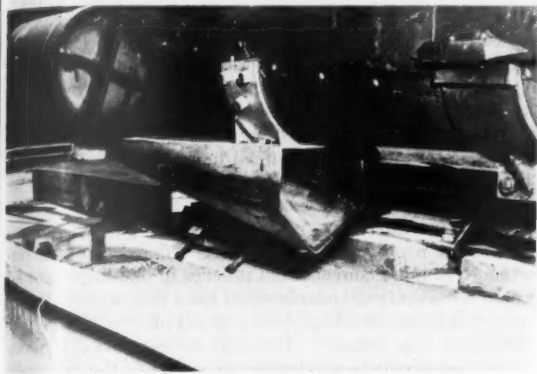


Fig. 7 Water brake mounted on test vehicle

trough of water has been placed between the rails for the last 2200 ft of track. The track slopes down approximately 6 in. in this interval. A metal scoop is rigidly attached to the bottom of the sled. At station 8000 this scoop enters the water. It is designed to pick up the water and turn it through 160°, thereby imparting the sled's kinetic energy to the water (Fig. 8). The configuration of the scoop and the rate at which it enters the water are designed to provide a constant decelerating force sufficient to stop the sled at station 9800 for any initial velocity.

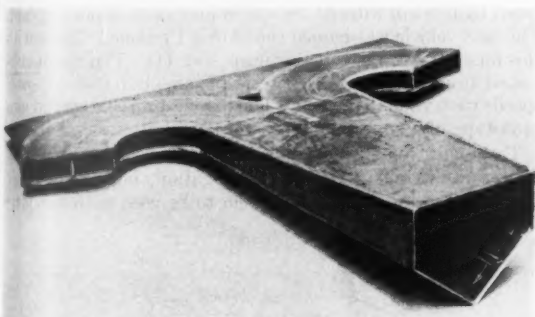


Fig. 8 Water brake

One hundred and forty tests have been conducted with this vehicle. Speeds up to 600 mph have been attained.

This sled is now being modified for a program of tests to determine the aerodynamic coefficients (drag, lift, and pitching moment) of a large ordnance-type store. For these tests, the capsule at the top of the sled has been removed and replaced with full scale model of the store. This store is mounted above the sled through a box type strain gage sting of the type commonly used in wind tunnels. Values of the aerodynamic forces as functions of velocity will be recorded continuously on a magnetic tape recorder.

Fig. 9 shows another test vehicle which is currently being used to study parachute performance at sonic and supersonic speeds. The sled is powered by a gas pressurized, liquid propellant rocket motor. The fuel for this motor is a mixture of 75 per cent ethyl alcohol and 25 per cent water. The oxidizer is liquid oxygen. A gaseous mixture of nitrogen and helium is used for pressurizing the fuel system. The thrust is adjustable from 30,000 to 50,000 lb. The maximum duration at 50,000 lb thrust is approximately 4½ seconds. The main structural member of the sled is the high pressure nitrogen

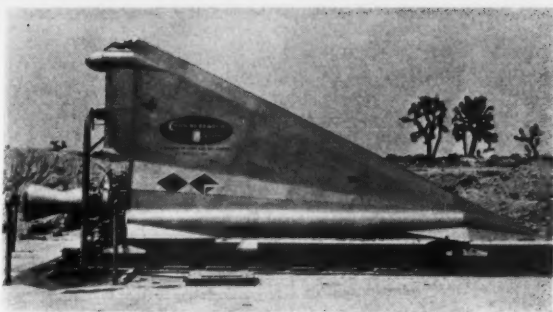


Fig. 9 Supersonic parachute test vehicle

tank which is 10 in. in diameter and fabricated from ¼-in. SAE 4130 heat-treated steel (see Fig. 10). It is charged with a mixture of gaseous nitrogen and helium to a pressure of 3000 psi. Box frame cross-members are welded to each end of the nitrogen tank. The alcohol and liquid oxygen propellant tanks are 12 in. in diameter. They are fabricated from ¼-in. 61S-T aluminum tubing and are attached to either side of the nitrogen tank with fixed supports at the rear. The front supports permit longitudinal movement to accommodate growth or shrinkage of the tanks. The fuel and oxidizer tanks are charged to a maximum pressure of 600 psi.

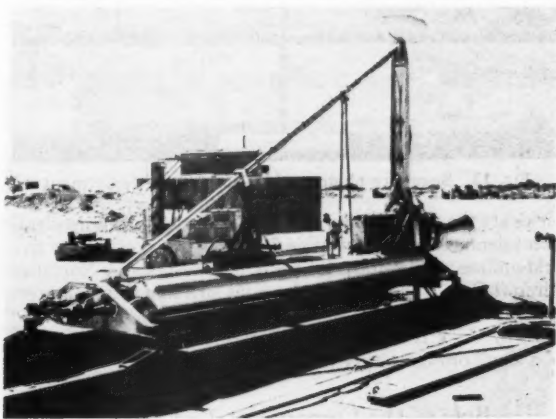


Fig. 10 Basic structure and power plant of supersonic parachute test vehicle

The thrust chamber body is of conventional design with a chamber to throat area ratio of 2:1 and an expansion ratio of 3.65:1. It is fabricated from SAE 1020 mild steel and weighs 259 lb. The injector is of the triplet impinging type and weighs 62 lb. Ignition is obtained with two integral alcohol-oxygen igniters which are fired by aircraft-type spark plugs.

All valves are controlled pneumatically by 750 psi nitrogen supplied through solenoid actuated valves. The tubular diagonal support between the front of the sled and the parachute attachment point also acts as an auxiliary source of high pressure nitrogen for the control system.

The skin of the sled was fabricated from 0.064-in. thick 75S-T aluminum suitably braced to carry the aerodynamic loads.

The instrumentation, photographic equipment, magnetic tape recorder, and water brake previously described were used on this sled. Over sixty tests have been conducted with this vehicle. On many of these tests, supersonic speeds have been attained. A typical test with a FIST Ribbon parachute is shown in Figs. 11 and 12.

Many other rocket powered track type test vehicles are now being operated on the free air test facilities at Holloman Air Force Base, at Edwards Air Force Base, and at NOTS, Inyokern, Calif. Typical of the varied applications of these ve-

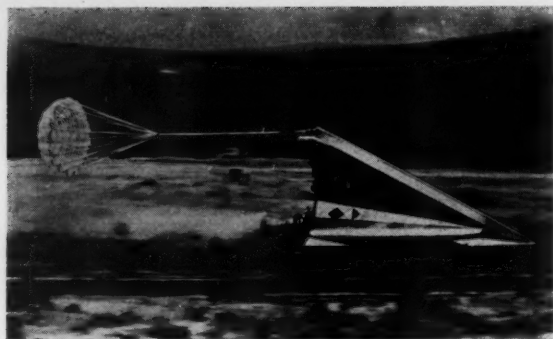


Fig. 11 High speed parachute test

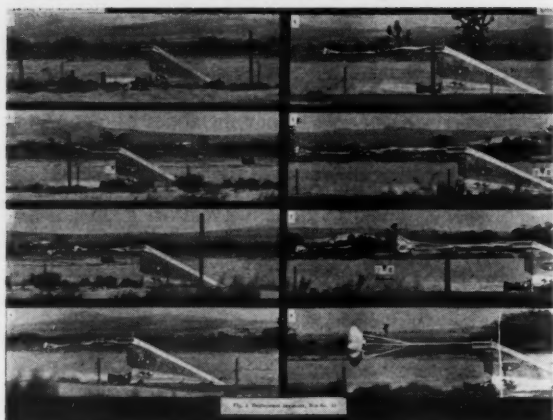


Fig. 12 Sequence photographs of parachute deployment

hicles are the sleds used by Col. J. P. Stapp to test the deceleration tolerances of human subjects; the sleds, powered by five acid-aniline motors and used for aeroelastic tests by Northrup Aircraft Corp.; those built by various airframe manufacturers for full scale tests of ejection seats; and the 30-ft high sled built by Cook Research Laboratories for full scale drop tests of a small ordnance-type store.

### Role of the Component Test Missiles

Let us now consider the role of the component test missiles. Several such vehicles have been used in connection with two development programs being conducted by Cook Research Laboratories Division of Cook Electric Company.

In the first program, brake-type parachutes have been tested on a track type facility as part of a program to develop parachute recovery systems for missiles and target aircraft. It is well known that parachute opening characteristics and particularly opening shock forces vary with altitude (more accurately, with air density). Also, the sled tests have been conducted with a track borne vehicle whose stability is assured by its very nature. It was, therefore, necessary to validate the results of these track tests at representative altitudes and with a vehicle which provided more realistic stability characteristics.

The objective of the second program was to evaluate physiological factors associated with emergency escape from supersonic aircraft. As a first step in this program, it was necessary to determine the actual values of deceleration, wind blast, and angular velocity to which a pilot would be subjected in an ejection seat escape from a supersonic aircraft. Dynamic tests in a supersonic wind tunnel would be very difficult to perform and would require extremely small models. Questions of dynamic similarity and the need for dimensional scale factors would seriously reduce the reliability of results. Sled tests could not be used for this type of testing because in-

sufficient altitude would be available after ejection to provide adequate information about the dynamic response and trajectory of the ejection seat, and similarly the effects of altitude cannot be determined. Factors involving economy, availability of the aircraft, and the feasibility of the tests make the use of actual supersonic aircraft for these tests impossible.

In both the above programs there was a definite requirement for a component test missile which could be economically fabricated, which would be capable of supersonic performance, and which would provide a method of obtaining maximum information about equipment performance. In addition, the test vehicles should be reusable. For example, if successful parachute recoveries could be accomplished, the required test programs could be performed more economically. Test missiles which were designed and built to satisfy these requirements are described below.

It was immediately apparent that the performance required from a component test vehicle differed from the performance of more conventional missiles in one important aspect. A conventional missile either has a long period of power flight or it is required to glide in a power off condition at high speed for a long distance. It is only necessary that the component test missile be accelerated to speed and the equipment under test be allowed to function. The operation of the final recovery system then can be initiated. Since the time of flight is short, the need to minimize missile drag (and, to a lesser degree, missile weight) becomes much less important than in conventional missile design. High drag, in fact, becomes an advantage after the test equipment has functioned because it simplifies the recovery of the missile. This principle was used in the design of component test vehicles for both the above applications.

Two different missiles have been used for parachute validation tests.

The first vehicle was designed to attain a high subsonic speed by free fall without the use of an auxiliary power plant. This test vehicle was named the Skokie I missile. Tests with this missile served two major purposes: (1) The results obtained from a series of sled tests of parachutes at subsonic speeds were validated. (2) The Skokie I missile served as a prototype of a supersonic powered parachute test missile.

Tests with Skokie I provided information about the configuration, stability, method of operation, instrumentation, and the parachute recovery system to be used with the later powered vehicle (Skokie II).

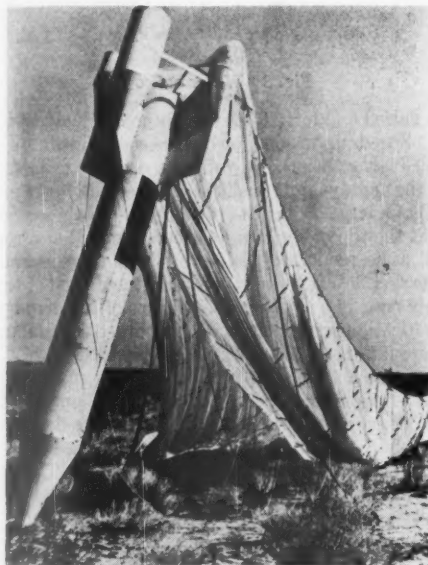


Fig. 13 Subsonic free flight test vehicle

The Skokie I missile is shown in Fig. 13. Specifically, it was designed to attain a maximum speed equivalent to a dynamic pressure of 400 psf at approximately 12,000 ft. The required performance was obtained by carrying the missile aloft in the bomb bay of a modified B-29 to an altitude of approximately 30,000 ft. At this altitude it was released and allowed to free fall to the required test velocity. The test parachute was then deployed.

A tensiometer was located between the parachute and the missile to measure parachute drag force. The output of this tensiometer was recorded on a magnetic tape recorder located in the nose section of the missile. Outputs of pitot pressure gages and accelerometers were also recorded. Cameras located in the tail section of the missile provided information about parachute-opening characteristics and stability.

At the end of a 6-sec test period, the test parachute was released and an 11.9-ft FIST Ribbon or 9.0-ft Guide Surface Ribless first-stage recovery parachute was deployed. This parachute decelerated the missile to a speed at which an 88-ft diameter final recovery parachute could safely be deployed. Damage to the missile on ground impact is minimized by means of the spike landing shown in Fig. 14.

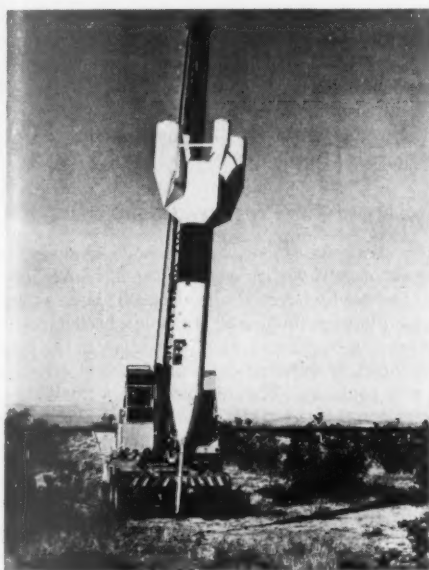


Fig. 14 Subsonic free flight test vehicle

There was no requirement to minimize weight in the design of this missile, so steel construction was used. The nose section was fabricated from  $\frac{1}{4}$ -in. plate with rib type stiffeners. Steel channels form the main structural members of the instrument and fuselage sections. All parachutes were located in the pods at the tip of the fins. The weight of the missile was approximately 2500 lb.

This type of missile was flown nine times and eight successful recoveries were accomplished. On the unsuccessful recovery the missile was completely demolished by ground impact, but the magnetic tape was recovered intact and provided a complete record of parachute performance. Data previously obtained from sled tests were generally validated as a result of this missile drop test.

A supersonic parachute validation missile also has been fabricated and is in the process of being tested. This missile is called the Skokie II. It is powered with three 11,000-lb thrust JATO units, each of which provides a total impulse of 25,000 lb sec. This missile will also be air launched and, with a dry weight of 3000 lb, should attain a speed of Mach 2. A parachute recovery system similar to that used on the Skokie I is incorporated in the vehicle (see Fig. 15).

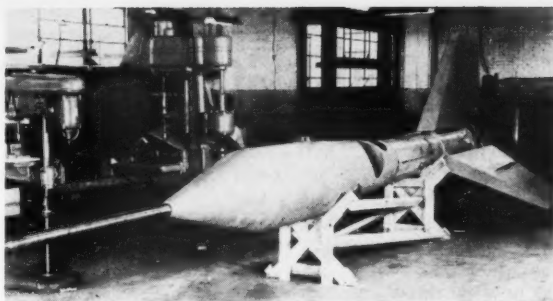


Fig. 15 Supersonic free flight parachute test vehicle

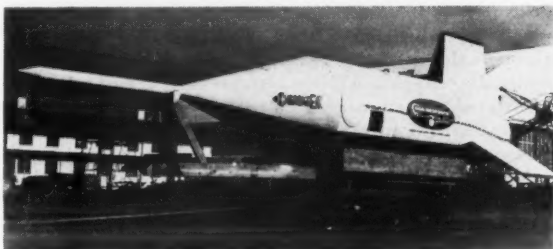


Fig. 16 Rocket-powered component test vehicle—Cherokee

The component test missile which was built to study the flight characteristics of ejection seats at supersonic velocities is shown in Fig. 16. This vehicle was named the Cherokee missile.

It was designed to permit testing of a full-scale standard Air Force ejection seat with a simulated pilot at speeds up to Mach 1.5 at 20,000-ft altitude. The ejection seat and associated instrumentation weigh approximately 300 lb and required a cabin section at least 54 in. in diameter.

A solid propellant missile JATO unit providing approximately 50,000-lb thrust for three seconds was chosen for the missile power plant.

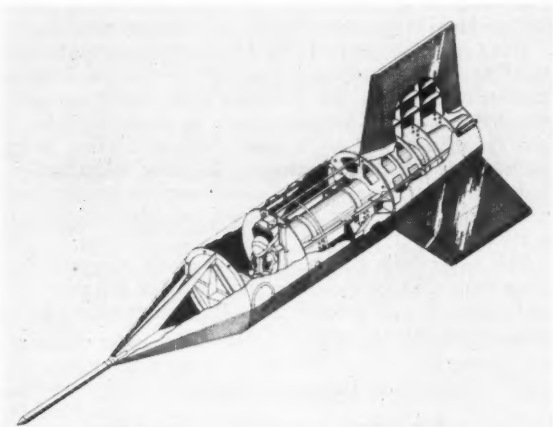


Fig. 17 Cherokee construction details

Details of missile construction are shown in Fig. 17. The missile nose section was designed in steel plate to balance the heavy JATO unit at the aft end of the missile and to maintain the missile CG at a station consistent with stability requirements. The cabin section was welded construction using 61-S-T aluminum alloy. The JATO unit itself provided the main structural strength of the aft tail section. Because drag reduction was of secondary importance compared with ease of fabrication, no attempt was made to counter-sink rivets or recess fittings.

First and second stage recovery parachutes were stored in

(Continued on page 466)



# Generation of High Gas Pressure Through Hydraulics

JULIUS KENDALL<sup>1</sup>

Greer Hydraulics, Inc., N. Y. International Airport, Jamaica, N. Y.

This paper covers the development of a type of portable ground handling equipment capable of furnishing high pressure gas in ranges up to 6000 psi, and describes the means by which this high pressure gas is developed through the utilization of high pressure variable volume hydraulic pumps and Greer high pressure transfer barriers connected to plenum chambers to store the high pressure gas until required. The transfer barrier acts as a means of separating the high pressure oil (which is used to develop the high gas pressure) from the actual gas itself in order that, in the development of this pressure, the gas is not contaminated in any way by oil or moisture of any type. By a predetermined cycling arrangement the desired volume and rate of gas desired can be controlled to satisfy the operator's requirements based upon selection of the capacity of the pump and the number of transfer barriers employed.

## Introduction

**D**URING the past several years, rocket engines have grown both in scope of application and in volume of units being produced. Rocket engines have now reached the state of development and application requiring a "limited" production status in several commercial organizations. No longer do our scientists and engineers toil over one unit in order to obtain a reasonable percentage of anticipated performance. Today, rocket engines, by comparison, can be readily designed, tested, and produced in quantities with a high degree of reliability.

Due to this increase in the use of rocket engines, make-shift handling and test equipment, designed to perform at a limited number of occasions, are no longer economically or operationally feasible. Equipment must be made available to perform reliably and continuously without coaxing or frequent "between run" overhauls. In order to further expedite the handling and use of rocket engines, the necessary ground equipment must be designed for optimum safety to personnel.

This paper deals with one particular type of ground handling equipment for rocket engines. That is, the generation and supply of high pressure gas. By high pressure gas, we mean pressures in the range of 6000 psi.

## Pumping Fuels

Many rocket engines presently in use must employ a high pressure gas system with which to drive or pump the rocket fuels into the combustion chamber. In certain cases, pumps are employed, but these are limited due to the severe corrosive action of present-day fuels on the pump material. Therefore, the generation of continuous and adequate supply of high-pressure 6000 psi gas for propulsion systems used in missiles, sleds, or other test vehicles, presents an important problem. The problem is further complicated by the requirement that the gas be moisture and oil free at this elevated pressure to eliminate danger of chemical reaction be-

tween the contaminants in the gas and the fuel on which work is being performed.

## High Pressure Compressors

High pressure compressors have been developed in the past and are in use in numerous applications in the general technical industry. However, this equipment has its limitations. Almost all high pressure compressors are positive displacement piston-type units. This equipment must be lubricated with either water or oil. Therefore, the end product of compression requires elaborate drying and filtering to remove the contaminating water and oil. Doing this at a pressure range of 6000 psi is both difficult and costly. Secondly, the compressors themselves become very costly as the amount of delivered gas required at 6000 psi increases. Since abundant lubrication cannot be used, costly overhaul of these compressors is required at frequent intervals.

## Gas Pressure Booster

The disadvantages in present methods of generating high pressure air or inert gas gave rise to the development of a gas pressure booster by Greer Hydraulics. This pressure booster is contained in a medium-sized portable wheeled trailer to permit use of this equipment at any location in the field. The unit is completely self-contained except for electric power for motors and controls. The advantage of a portable pressure booster at a missile test station is evident. First, the small number of missiles which are fired over a period of time would not justify the permanent installation of costly equipment at one site. Long extensions of high pressure lines are not practical from the point of view of pressure drop, possible leakage, and cost. Therefore, a unit of this type can be utilized on a scheduled basis for a wide variety of applications at any desired location on the test station. Large multistage compressors would not lend themselves to this kind of field application.

This pressure booster utilizes hydraulics to compress and pump gas. The booster system consists basically of: one, a source of low pressure gas which can be commercial bottles or

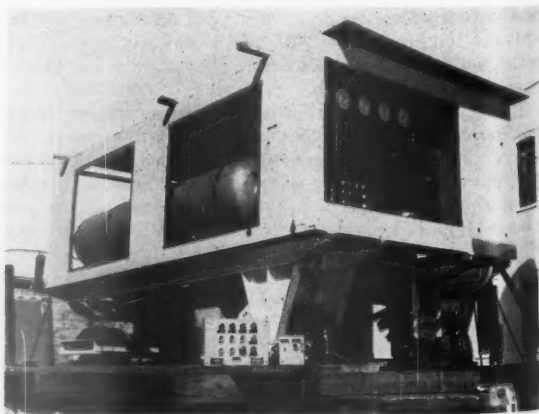


Fig. 1 Over-all photo of gas booster unit

Presented at the ARS Fall Meeting, El Paso, Texas, September 22-24, 1954.

<sup>1</sup> Vice-President. Mem. ARS.



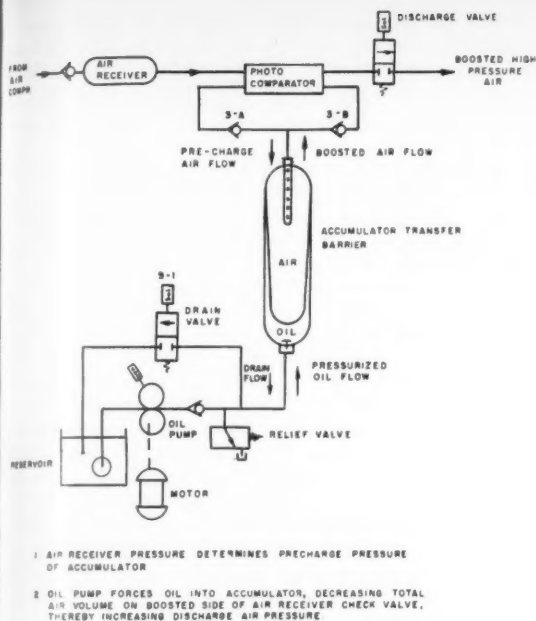


Fig. 2 Schematic of booster circuit

a low pressure mechanical compressor; two, the hydraulic transfer barrier; and three, a high pressure receiver. Although the application has been used in the past, this machine represents a large step in refinement.

The particular function of this unit is to take a high input of compressed gas and deliver it more highly compressed and at a much higher rate than is possible of using the existing equipment. A battery of accumulator-type transfer barriers serve as the compression cylinder for the gas. The system also includes a high pressure hydraulic pump, electric motor drive, oil reservoir, pressure activated switches, solenoid valves, float switches, air receivers, filters, and remote control equipment.

### Operation of Booster Unit

The operation of the booster unit is as follows: Prior to the desired time of operation, a low pressure compressor (3500 psi) is employed to charge up two receivers and a battery of transfer barriers with 3500 psi oil and moisture free air. All transfer barriers are completely drained of oil to assure complete filling with pressurized air or inert gas. After the booster unit has been completely charged and the missile is ready for charging, the booster unit gas delivery line is connected to the missile. Upon opening the gas delivery valve, there is a brief period of flow from the booster unit to the gas receiver in the missile. In this particular application, pressure equalizes at approximately 2400 psi. Hydraulic oil is then pumped at a predetermined rate, set by the side of the oil pump, into the oil side of the transfer barriers. The displacement of the bag by the oil, compresses the gas contained therein. The compressed gas is forced out of the transfer barrier, through a system of check valves, into the missile high pressures receiver.

### Transfer Barriers

The transfer barriers are filled to approximately 80 per cent of their capacity. At this time, a low level float switch in the oil reservoir shuts off the hydraulic pump and indicates through a light on the panel that the first cycle has been completed. Thereupon, the operator actuates solenoid operated valves to release the oil back to the reservoir. The high pressure gas retained in the almost collapsed bag inside the transfer barrier expands. The expanding bag forces the oil back

into the reservoir. During the expansion of the gas in the bags, the pressure drops below the pressure in the two large gas receivers. Gas from the receivers then flows into the transfer barriers as part of the recharging cycle. The high level float switch on the oil reservoir introduces orifice flow restriction in the oil return line just prior to complete emptying of the transfer barriers. This introduces a sharp reduction in the rate of flow discharge to prevent impact of the bag against the valve in the transfer barrier shell. After return of the oil to the oil reservoir, the drain valves are closed and the hydraulic pump restarted to repeat the second, and subsequent, compression cycles.

The transfer barriers employed in this system are, in essence, a modified accumulator. In the Greer Accumulator there is a rubber bag which acts as a separator between two different fluids. The bag is usually precharged with nitrogen gas to a pressure governed by the particular application. Oil is introduced into the accumulator by the pressure in the hydraulic circuit. The precharge is determined so that the accumulator will contain approximately equal volumes of oil and gas during normal operation of the hydraulic system. The gas, which is compressible, varies in volume which pressure surges in the hydraulic line and hence acts as a shock absorber or pulse dampener.

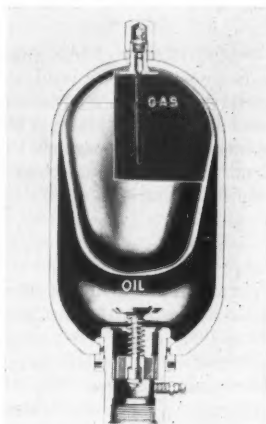


Fig. 3 Cross section of transfer barrier

In the transfer barrier application, either fluid is subjected to flow by the increase of pressure of the opposing fluid. In the pressure booster application discussed in this paper, high pressure gas flows into the bag by relieving the oil circuit to atmospheric pressure. On the upstroke of this cycle, high pressure oil is introduced, compressing and forcing the gas out of the bag through the use of appropriate check valves. This alternating flow or cycle is repeated by relieving the driving oil pressure and allowing recharge flow of the air. Hence, the action of a piston in a cylinder is duplicated except that there is no requirement for lubrication or physical contact of the mediums involved to cause a contamination in the fluid being pressurized.

Modification of the accumulator is necessary in order to adapt it to this application. First, provision must be made for permitting the flow of gas in and out of the bag. Since transfer barrier action more or less collapses the bag, provision must be made to prevent extrusion of the bag out of the air port. A long perforated tube is employed to provide the necessary port area and still prevent extrusion of the bag. High flows encountered in the transfer barrier application require that the oil poppet valve be redesigned to function under more severe hydrodynamic conditions.

### Control Panels

The schematic that has just been discussed represents a

simplified explanation of the basic system. In the actual equipment, a complete local control panel, a complete remote control panel, and a missile operator's panel are provided.



Fig. 4 Missile operator's and remote control panel

Control of any phase of delivery of high pressure gas to the missile can be made from either the local or remote control panel. This necessitates remote transmission of pressure and flow information and an intricate system of solenoid operated valves. The simplified Missile Operator's Panel permits the transmission of final instructions for pressure and flow adjustment from the missile firing station.

### Compressor

This type of compressor has a definite sphere of application. Examination of the characteristic adiabatic or isothermal "pressure vs. volume" curves for the compression of air indicates that it is impractical to use this equipment for initial pressures much lower than 2000 psi. Boosting gas pressure from 2000 psi to 6000 psi is a range of practical application for this system. Also, compression ratios greater than 4:1 are not practical with this type of compressor. The slope of the "pressure-vs.-volume" curve in the low pressure range requires a considerable number of compression cycles to obtain the high limit of pressure. A higher pressure range of the pressure-vs.-volume curve requires fewer compression cycles to obtain the high pressure limit of this particular range. A greater number of cycles, hence, a greater cycling rate, would yield a compression cycle approximating that of an adiabatic pressure-volume condition. This would cause a rapid rise in the temperature of the gas (and the bag) and would lead to early failure of the transfer barrier bags. However, a lower cycling rate would yield a pressure-volume compression curve closely approximating that of an isothermal condition. In

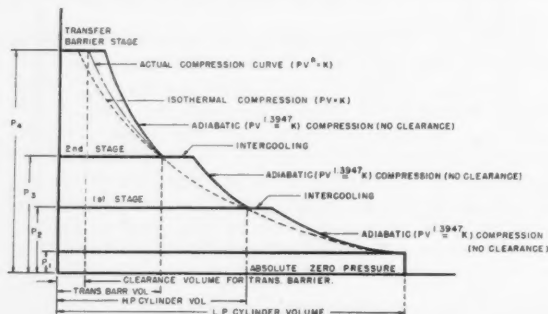


Fig. 5 Pressure-vs.-volume curve for gas booster system

this case, temperature of the gas and the bag would be maintained by dissipation of heat by the oil and by radiation from the hardware in the circuit. Hence, the life of the bags is extended to a reasonable period.

Since high pressure gases are involved with this equipment, the safety problem must be emphasized. Failures of high pressure gas systems are dangerous, since gas retains its high energy level for a period of time following fracture of a component. In a liquid system, flow through a fracture would cause a quick drop in the energy level of the fluid. Needless to say, the use of compressed air in a system employing hydraulic oil appears to be a most dangerous combination of conditions. To eliminate this danger, it is important that no oil be permitted to mix with the high pressure air. This is prevented in the pressure booster by the addition of an electronic Photo-Comparator Unit with slaved solenoid shut-off valves in the circuit. The Photo-Comparator, by means of a photo cell, compares the delivered air flow with the supply flow in the booster system. The supply air flow is considered as a reference since this air has been dried and filtered immediately upstream of the Photo-Comparator. An oil leak causing a few oil droplets to enter the delivered air supply would cause a discrepancy in the color or density of the air. This discrepancy is immediately detected by the Photo-Comparator which shuts off the supply of oil-contaminated air to the rest of the system.

Standard acceptable fittings and other circuit components are not readily available for use in 6000-psi gas pressure equipment. Therefore, it was necessary to subject all equipment to laboratory burst pressure tests. By certifying all high pressure equipment incorporated in the pressure booster, a high degree of safety is claimed. Safety blowout diaphragms, relief valves, and other generally accepted safety devices are employed wherever necessary.

This particular gas pressure booster has been designed for flows up to 660 SCFM at 6000 psi pressure. The initial performance requirements were as follows:

- 1 Compressor required to fill a 6-cu-ft bottle with 6000 psi of air in five minutes.
- 2 Compressor required to fill a 20-cu-ft bottle with 3500 psi of air in twenty minutes.

As already related, the system was designed specifically for use with atmospheric air. However, the identical system can be employed for boosting pressure of gases such as nitrogen or helium. Other systems have been designed specifically for nitrogen and helium applications. In one particular application for helium gas, the number of transfer barriers and cycling rate of compression resulted in an adiabatic condition. In this case, it was necessary to employ heat exchangers in the hydraulic and gas circuits to maintain control of the fluid temperature. Dissipation of heat from the gas and from the oil is then a continuous process which prevents any significant rise in the delivered high pressure gas.

### Compressor With No Mechanical Parts

This type of compressor is interesting from several points of view. High flows at high pressures can be obtained in comparably short intervals of time. The cost of this equipment is in the neighborhood of one third of that of existing mechanical compressors. The number of mechanical moving parts requiring replacement are at a minimum. This equipment requires only periodic replacement of rubber bags, which is an inexpensive and simple operation. Another important point is that the pressure booster can be used for atmospheric air one day, nitrogen gas a second day, helium on the third day, and so on—with no modifications or component changes necessary in the system. A final and most important feature

(Continued on page 467)

# The Application of Radiation Measurement Techniques to the Determination of Gas Temperatures in Liquid Propellant Flames

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Radiation temperature measurements were made throughout the flame developed within an open-tube combustor using liquid oxygen and a heptane-turpentine mixture as the reactants. The temperature measurement technique applied was the two-color radiation method. Observed absolute temperature measurements did not agree with theoretical equilibrium flame temperatures in most cases. The results can be explained by the existence of one, or a combination of three conditions within the chamber: (a) gas temperature stratification; (b) lack of thermodynamic equilibrium; and (c) optical effects such as selective absorption and scattering of light. These conditions limit measurements to qualitative observations concerning absolute temperatures as a function of flame position and time. In general, gas stratification effects were found to become more pronounced as the flame was made increasingly fuel rich. Flame temperatures were found to vary inversely with the radiation intensity, and fluctuating temperatures were found to exist at a frequency compatible with an acoustical mode of the chamber, even when sound oscillations of the same frequency were not apparent. Very high frequency oscillations of temperature, emissivity, and radiation were also found.

## Introduction

THE application of radiation-measurement techniques to the determination of gas temperatures in the flame resulting from liquid propellant reactions has recently been under investigation at the NACA Lewis Flight Propulsion Laboratory. Such techniques are desirable in rocket combustion and injector design studies because they permit the study of thermodynamic conditions in a flame zone without disturbing the flow and without the necessity of maintaining a probe in the chamber.

Gas temperature measurements by radiation techniques are usually variations of the sodium line reversal and various carbon radiation techniques. The theories of these techniques are presented in (1, 2, 3, 4, 5).<sup>2</sup> Their previously reported application to rocket engines is presented in (5, 6, 7, 8).

The radiation temperature measuring techniques referred to above are based on the characteristics of thermal radiation, which, throughout the visible spectrum, are adequately described by Wien's radiation equation

$$J_\lambda = C_1 E_\lambda \lambda^{-5} e^{-C_2/\lambda T}$$

where

$J_\lambda$  = monochromatic radiation intensity

$E_\lambda$  = monochromatic emissivity of radiator

$\lambda$  = wavelength

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<sup>1</sup> Aeronautical Research Scientist.

<sup>2</sup> Numbers in parentheses indicate References at end of paper.

$T$  = true temperature of radiator

$C_1$  and  $C_2$  are known constants

With the use of various techniques of measurement described in the literature, radiation temperature may be determined from observation of the radiation intensity of a hot gas or body. In the use of sodium methods, sodium is added to the flame, and sodium atoms serve as the radiating particles. In the use of carbon radiation methods, the continuous radiation of carbon particles produced during the combustion of the fuel is utilized.

This paper presents information gained with the use of the two-color method of temperature measurement. The theory, presented in (1), depends on carbon particle radiation. The emissivity of carbon particles as a function of wavelength was empirically determined on amyl acetate flames as reported in (1), and the theory then checked experimentally. These results should be applicable to any system which is in thermal equilibrium where the observed radiation is only from carbon particles similar to those reported in (1). Any spectral distortion of the supposed carbon continuum at the wavelengths being observed, makes the theory inapplicable. Such distortion is likely to occur in certain cases in liquid propellant flames. Where it was believed to occur and how it affects temperature measurements are discussed later.

The two-color method with instrumentation refinements was applied to the exhaust zone of a liquid hydrocarbon-liquid oxygen flame, as reported in (6). The fuel was a mixture of turpentine and heptane which gave a flame of high carbon luminosity. In the work reported here, the investigation was extended to the study of temperatures in the chamber as well as the exhaust zone.

The reduction of radiation intensity records to temperatures often resulted in determined temperatures which were higher than theoretically possible for equilibrium conditions. However, qualitative information concerning the combustion process was gained from relative temperature measurements and fluctuating temperatures and spectral emissivities throughout the flame. Both the limitations of this method of temperature measurement and the qualitative information gained with its use are discussed in this paper.

## Combustion System

Radiation temperature measurements were made of a flame developed in an open tube combustion chamber of 2-in. diam and 27-in. length. Propellants were introduced into the chamber through a triplet injector that caused two oxidant streams to impinge on a fuel stream at a total flow rate of 0.2 lb/sec. Propellants used consisted of a heptane-turpentine mixture as fuel with liquid oxygen as the oxidizer. The combustor was operated over an oxidant-fuel weight ratio range of 1.2 to 2.4. A chamber pressure rise of less than one atmosphere was encountered during operation. In order to aid in identifying unstable combustor operation, a crystal

microphone was placed near the exhaust zone and a record of its output signal was obtained.

A photograph of the combustion chamber is shown in Fig. 1.

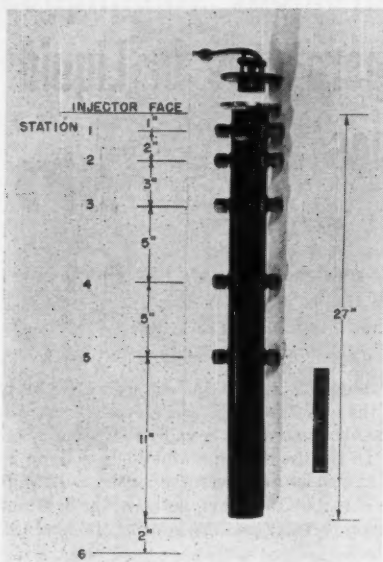


Fig. 1 Combustor and injector, showing station locations

Window ports were located at five axial positions in the chamber. The positions, stations 1 through 5, respectively, were 1, 3, 6, 11, and 16 inches from the injector face. Two-color temperature determinations were made at the five stations within the combustor and at station 6 located at the exit of the chamber.

### Temperature Measurements

In applying the two-color method of temperature measurement, radiation intensities at two distinct wavelengths are measured. The intensities are related to temperature and emissivity by Wein's equation and experimental emissivity relationships as reported in (1).

This study used a spectrometer with photomultiplier tubes located to receive radiation from the combustion gases at three wavelengths (Fig. 2). Wavelengths of 6400 Å and 4850 Å were generally used. A third wavelength of 5500 Å provided a check of results by allowing two independent temperatures to be obtained simultaneously. The wavelength bands were 30 Å at 4850 Å and 100 Å at 6400 Å. These band widths compensated for the spectral response of the tubes. A broad ribbon tungsten lamp served as the standard com-

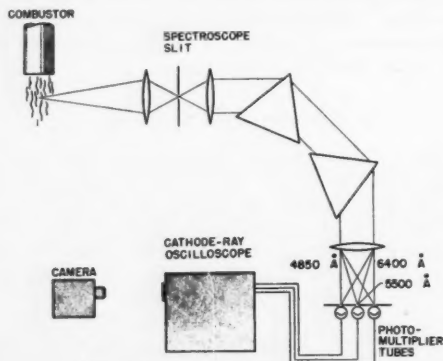


Fig. 2 Schematic drawing of two-color temperature apparatus

parison source. The spectroscopy aperture was always filled by the observed radiation.

Data were recorded from oscilloscope traces of the tube output signals. Photographic records of 2-4-millsec duration were originally obtained. Later improvements in the recording system allowed full runs of 3-sec duration to be recorded. Radiation records were reduced to temperature and emissivity using a point-by-point analysis; a time averaged mean of the points represented the average temperature. The maximum frequency at which data were considered to be reliable was 3000 cps. This limit was due primarily to the electronics in the recording system.

## Results and Discussion

### Average Temperature Measurements

Average temperatures determined in the exhaust zone (station 6) are shown in Fig. 3. Temperatures determined within the chamber at stations 4 and 5 are shown in Fig. 4. At all stations, temperatures were measured that were above theoretical equilibrium temperatures. The measured temperature as a function of distance along the chamber axis for an O/F of 2.4 is shown in Fig. 5. The departure of the measured temperature from the theoretical equilibrium temperature is evident in the figure. At stations 1, 2, 3, and 4, average determined temperatures were above the theoretical equilibrium temperature at stoichiometric oxidant-fuel ratio of 3.66.

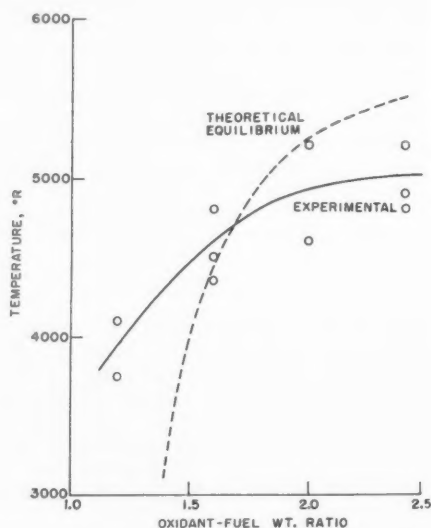


Fig. 3 Average exhaust-zone temperature measurements (station 6)

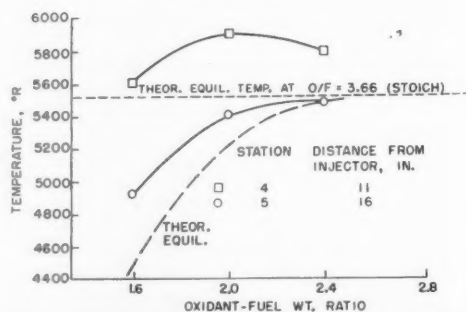


Fig. 4 Variation of temperature with oxidant-fuel weight ratio obtained with two-color measurements



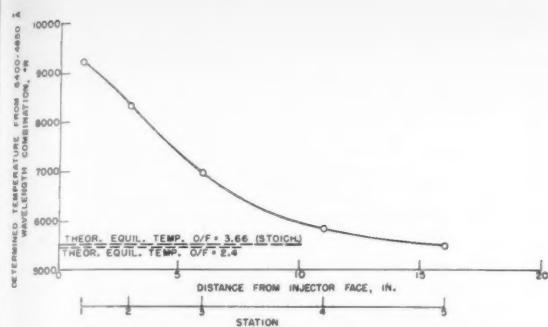


Fig. 5 Variation of average determined temperature with distance from injector face for oxidant-fuel ratio of 2.4

The determination of two-color temperatures which are high with respect to theoretical temperatures indicates that the two-color method is not capable of measuring true average gas temperature in a highly complex flame such as the one investigated here. The value of the method lies in qualitative information which may be gained concerning combustion and its completion along the axis of the combustor, and in the determination of the relative magnitude of rapid temperature variations.

The inability of this instrument to measure true gas temperatures in a highly heterogeneous flame may be attributed to one or a combination of three effects: (a) Gas temperature stratification; (b) thermal equilibrium or equi-partition of energy, not achieved; and (c) optical effects, such as light scattering and selective absorption due to liquids and small particles in the combustion zone.

### 1 Gas Temperature Stratification

Generally, radiation measurements tend to average toward the highest temperature existing in a zone under observation. In the burner used in this study, gas temperature stratification most likely existed at all stations. However, its effect on average temperature determinations would predominate in the exhaust gases and stations far from the injector. At these points, there is the best chance for thermodynamic equilibrium to be established. Also optical effects should be minimized because most liquid droplets should be vaporized. Thus, the high temperatures observed at stations 5 and 6 (Figs. 3 and 4) may be considered likely to be caused by gas temperature stratification. These high temperature determinations are seen to be above the equilibrium temperature for the O/F ratio set, but not above the maximum theoretical temperature for the propellant combination investigated. Pockets of gas which burned at off O/F ratios, resulting in high local temperatures, may be disproportionately raising the average temperature read by the instrument. This effect is more noticeable at station 5 than at station 6.

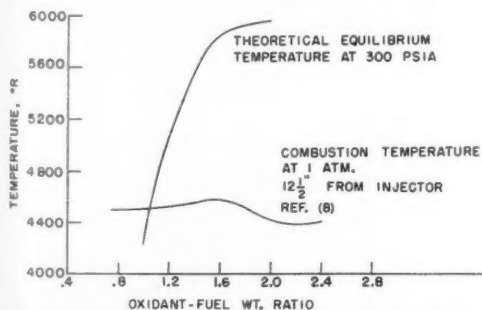


Fig. 6 Variation of combustion temperature with oxidant-fuel ratio for sodium line reversal method

Greater stratification at station 5 would be consistent with the observation. The effect also becomes more noticeable with decreasing O/F ratio, indicating that fuel rich flames require greater distances to burn and mix than do relatively lean flames with the propellant preparation system used in this investigation. Fig. 6 shows that a very similar effect was observed with the modified sodium D-line reversal method reported in (8). The data of Ref. 8 were obtained on a burner similar to the one used in this investigation, but alcohol rather than a hydrocarbon was used as the fuel. The use of alcohol instead of a hydrocarbon fuel is considered by the authors of this paper to have little effect on the qualitative results attributed to gas temperature stratification.

### 2 Lack of Thermodynamic Equilibrium

At stations 1, 2, 3, and 4, all measured temperatures were found to be even higher than maximum theoretical equilibrium temperatures (Figs. 4 and 5). Such high temperatures must include either, or a combination of, optical effects and a lack of thermodynamic equilibrium. At station 4, the persistence of liquid droplets and fog seems unlikely. Data taken with the sodium method (8), shown in Fig. 7, indicate that emissivity did not change between 6 and 12 inches from the injector face. These distances are analogous to stations

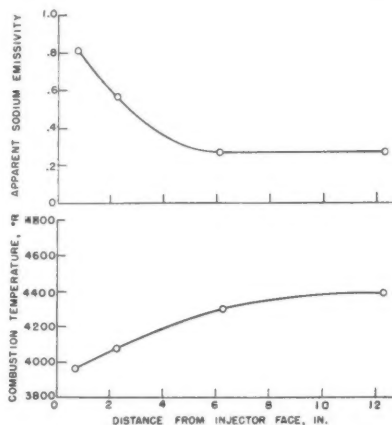


Fig. 7 Variation of apparent sodium emissivity and combustion temperature with distance from injector face for oxidant-fuel ratio of 2.4 (Ref. 8)

3 and 4 in the present investigation. In the sodium method described in (8), emissivity of the flame was measured as the flame absorption of tungsten radiation over a wavelength region slightly exceeding that of the sodium D-line doublet. This measurement required the assumption that flame absorptivity is equal to flame emissivity. This assumption is considered valid under strictly thermal conditions, but if absorbing and scattering particles exist in the light path, the assumption no longer holds true. In Fig. 7, it is seen that absorption did not change noticeably between stations 3 and 4, and so optical effects are considered not to be great in that region. Consequently, it is likely that the high two-color temperature determinations in that region may be attributed largely to lack of thermal equilibrium radiation. Gases not in thermal equilibrium would not radiate over a continuum in accordance with Wien's radiation equation as stated in the Introduction. Pockets of high temperature gases not in thermal equilibrium, or undergoing reaction, could cause high temperature averages. This effect may be exemplified by the existence of free radicals emitting spectral radiation in the region of the spectrum being observed. The existence of  $C_2$  and  $CH$  bands has been occasionally observed by the authors in a similar rocket engine operating at 300 psia chamber pressure. The existence of  $C_2$  bands would raise the radia-

tion intensity in the 5500 Å region above that which could be expected from thermal radiation. However, strong C<sub>2</sub> and CH bands do not appear in the 4850 Å and 6400 Å regions, and therefore their existence cannot explain high temperature determinations noted with the 6400 Å-4850 Å wavelength combination. Abnormal continuous radiation on alcohol-oxygen rocket engines has been previously reported in (5). The spectral distribution reported in this reference was such as to result in high temperature determinations, because of high radiation close to the 4850 Å region. The abnormal radiation observed in the blue region of the spectra reported in (5), did not seem to be a result of band spectra. At present, there is no definite explanation for the observed spectral intensity distribution, but it is interesting to note, in regard to the blue region of the spectrum, that spectral flame intensity is high with two different fuels; namely, alcohol as reported in (5), and turpentine-heptane mixtures reported in this paper.

As mentioned before, the existence of certain spectral bands in the region of the injector does not fully explain the abnormal temperature data. Such results lead to the supposition that these very high temperatures may actually exist at some points in the chamber. This condition would certainly not be one of thermodynamic equilibrium. However, it further suggests that the reaction may be taking a course not normally expected in engines of this type. It is conceivable that the reaction could progress as a series of small, high temperature detonations.

The authors have noted that engines with injectors designed to premix heptane and oxygen tend to explode. An injector of this type was studied with the use of streak photography. This study was designed to determine whether or not the explosion carried into the supposed premixed liquids. It was found that high velocity detonations (6500 to 10,000 fps) traveled through the mixture regardless of the propellant phase. Furthermore, the authors have observed that generally, whenever liquid phase mixing is approached, detonations on starts are likely to occur. This is true whether or not the mixing occurs before or after the propellants enter the chamber. It is logical to assume that detonations may occur during combustion as well. The triplet injector used in this temperature study approaches the condition of liquid mixing, and thus might promote detonations.

A burning condition of this type would, in part, account for the performance success of triplet injectors, for the occurrence of detonations in the chamber would greatly increase the energy release rate.

### 3 Optical Effects

Temperature data taken at stations 1, 2, and 3 indicate the possibility of optical effects in the chamber as well. In this region, the temperature rose to very high values. Temperatures determined at these stations are shown in Fig. 5. Fig. 7 shows that flame absorptivity for the sodium D lines rose from 0.3 to 0.8 as the injector was approached in the upper half of the chamber (8). A sodium line absorptivity of 0.8 was unlikely in view of the fact that D-line absorptivity was measured over a wavelength band exceeding that of the Na D-line doublet. In contrast to the high two-color temperatures near the injector, the modified sodium D-line method reports lower temperatures as shown in Fig. 7. This would be the result of little change in radiation level, with an increasing flame absorption. High absorptivity with resulting low temperatures observed with the modified sodium line reversal technique could be caused by absorption and scattering of light due to liquid particles in the region of observation. Such effects could also be responsible for the high temperatures observed with the use of the two-color method as discussed below.

Fig. 8 shows the change in radiation spectral distribution as a function of distance from the injector. As measurements

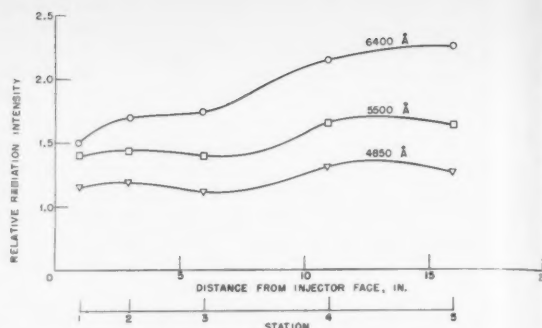


Fig. 8 Variation of average relative radiation intensity with distance from injector face for an oxidant-fuel ratio of 2.4

were taken progressively closer to the injector face, the intensity at 6400 Å was noted to drop in respect to the intensity at the other two wavelengths. Such a spectral distribution shift results in high temperature determinations as shown in Fig. 5.

Light scattered at an angle to an incident light beam by small particles is of different spectral distribution than that of the incident beam. Equations have been developed to describe this distribution for particles of various sizes. Rayleigh's law of scattering applies to particles smaller than the wavelength of light. For larger particles (of the order of light wavelengths) an equation developed by Mie applies. The equations are developed and defined in (9) and (10).

Assuming a limiting case where all the light viewed by the instrument is light coming from the lower part of the chamber and scattered at 90° to the incident beam, corrections can be made to the spectral distribution using the above equations. Such a correction could lower determined temperatures from 9000° to 5000° using the Rayleigh equation. Application of the Mie equation would lower determined temperatures from 9000° to 6500° R.

Selective absorption by liquid oxygen would also tend to lower the radiation at 6400 Å with respect to the radiation at 4850 Å. Liquid oxygen is known to exhibit a molecular band head in absorption at 6368 Å, which is in the region of the spectrum observed.

### Fluctuating Temperatures

Although it may be concluded from the above discussion that radiation temperature measurement methods are in-

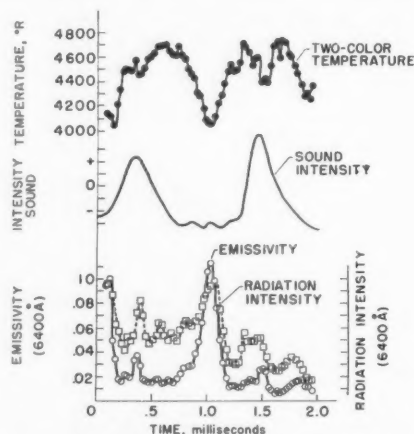


Fig. 9 Comparison of temperature, sound intensity, emissivity, and radiation intensity during audibly unstable operation. Oxidant-fuel ratio, 1.6

adequate for the determination of average gas temperatures within the chamber, the two-color method is of value in determining the relative magnitude of rapidly varying temperatures. Temperature, emissivity, and radiation variations were found to exist at all times during combustion. Variations of radiation, sound intensity, temperature, and emissivity with time in the exhaust zone (station 6) are shown in Fig. 9. This record was taken during audibly unstable combustion which was characterized by approximately 1000 cps variations in all parameters recorded. High frequency radiation variations on the order of 10,000 cps were also found to exist in the exhaust stream. It is likely that these very high frequency variations existed in temperature and emissivity as well, but the reduction of these data is questionable because the amplitude frequency response of the instrument was not flat above 3000 cps.

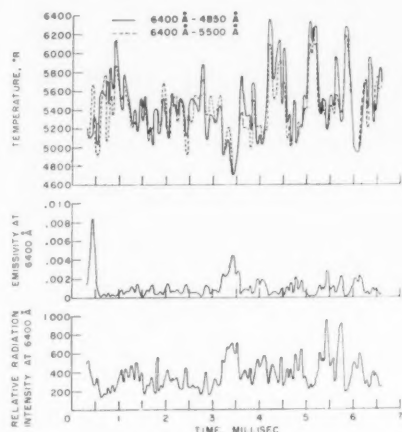


Fig. 10 Radiation, emissivity, and temperature in chamber at station 5, 16 in. from injector face. Oxidant-fuel ratio, 2.4

Similar variations in temperature, radiation, and emissivity were found to exist in the combustion chamber at all stations. Fig. 10 shows these variations at station 5 during audibly stable combustion. Independent temperatures determined from combinations of the three wavelengths used are presented. Reasonable agreement exists between the two temperatures, indicating that the relative magnitude of fluctuating temperatures was most likely reliable at station 5. It is evident from Fig. 10 that 1000 cps and higher frequency oscillations exist in all parameters. Consequently, the temperature variations exist at the same frequency as the sound during unstable combustion. These variations were found to exist even when the combustion was audibly smooth, as was the case when the data presented in Fig. 10 were recorded. As measurements were made progressively closer to the injector, it was noted that the high frequency components of radiation variation disappeared from the record, as shown in Fig. 11. No definite explanation of this observation is apparent, but it is possible that high frequency components of radiation were lost due to scattering and absorption close to the injector face. Such a condition could arise if high frequency radiation was the result of small pockets of hot or cold gases flowing by the windows. The radiation from these pockets would be absorbed and scattered in regions where liquid droplets or fog existed, but would appear further from the injector face where only gases existed in the chamber.

At all times, it was found that the monochromatic emissivity variations were 180° out of phase with the determined temperature variations. Radiation intensity variations were in phase with the emissivity variations. The effect has also been noticed on a low pressure fuel-air burner of the jet engine type as reported in (11). Whether or not true gas tem-

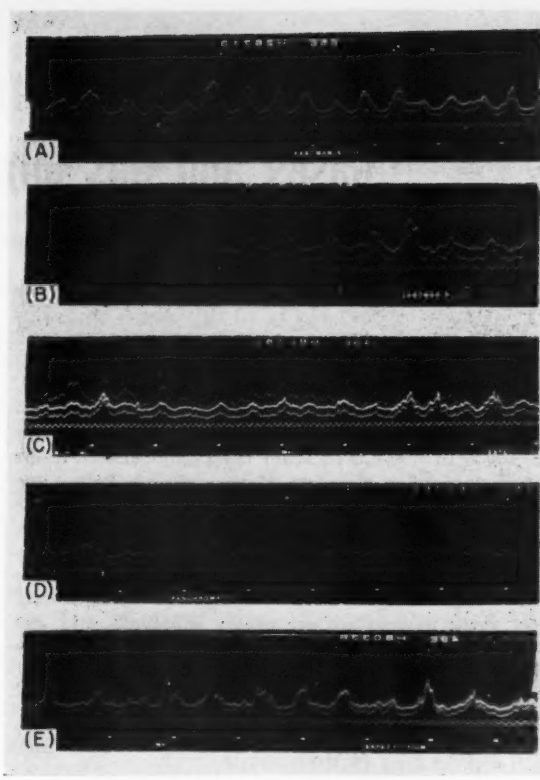


Fig. 11 Typical photograph of radiation traces at five stations in chamber. Rocket audibly unstable in (A), (B), and (E)

(A) Station 1, 1 in. from injector face. (B) Station 2, 3 in. from injector face. (C) Station 3, 6 in. from injector face. (D) Station 4, 11 in. from injector face. (E) Station 5, 16 in. from injector face.

peratures followed this trend is not known, since the temperatures determined by the two-color method cannot be relied upon to be indicative of the true gas temperatures in the flame.

## Conclusions

The effects of gas temperature stratification, lack of thermodynamic equilibrium, and optical effects prevented the reliable determination of average gas temperature with the two-color method in a liquid propellant flame. However, the application of this temperature measurement method to liquid propellant flames helps with a qualitative determination of completeness of combustion as a function of distance from the injector.

Variations in temperature and monochromatic emissivity of considerable magnitude were found to exist at an acoustical frequency of the flame tube, even when the instability was inaudible, indicating that energy differences exist at all times in the chamber at a frequency compatible with its acoustical characteristics. High frequency radiation variations were also observed in some regions of the chamber. These variations are believed to be perturbations in the combustion process. Whether or not they were coupled with the acoustical properties of the chamber was not determined.

Instantaneous temperatures determined with the two-color method were found to vary inversely with the radiation intensity. This inverse effect may not be true of the actual gas temperatures, but the results do suggest cautious interpretation of combustion photographs on the basis of luminosity.

(Continued on page 467)

# General Enthalpy-Temperature-Entropy Diagram for Ideal Gases and Gas Mixtures up to 5000 K

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A method was developed which permits the representation of the enthalpy and entropy of all bi- and triatomic gases and their mixtures in a single, general diagram. Large scale (21 × 43 in.) diagrams were drawn for the temperature ranges from room temperature to 5000 K and from 200 K to 1700 K which, for most applications, fulfill all functions of a complete Mollier chart. Values for individual gases are obtained by attributing to each gas two parameters. For the three most common gases, N<sub>2</sub>, H<sub>2</sub>O, and CO<sub>2</sub>, these parameters are constants and have simple numerical values. For the other bi- and triatomic gases, as well as for many monoatomic and higher atomic gases, these parameters are nearly constants over the major part of the temperature range. For gas mixtures composed chiefly of bi- and triatomic gases, constant parameters can be assumed in almost all practical cases.

## Nomenclature

$A$	= nozzle area, ft <sup>2</sup>
$b$	= parameter used by Stodola for general equation for molar specific heat (Ref. 3)
$\beta$	= parameter used by Lutz for general equation for molar specific heat (Ref. 4)
$C_p$	= molar specific heat for constant pressure, cal/mole-°K <sup>2</sup>
$C_v$	= molar specific heat for constant volume, cal/mole-°K
$c_0, c_1, c_2$	= general functions of specific heat, cal/mole-°K
$\gamma_1, \gamma_2$	= specific heat parameters for construction of general specific heat diagram
$\gamma_1, \gamma_2$	= parameter functions of particular gas fulfilling general equation for specific heat
$E$	= relative molar internal energy = $\int_{298.16}^T C_v dT$ , cal/mole
$H$	= relative molar enthalpy = $\int_{298.16}^T C_p dT$ , cal/mole
$e_0, e_1, e_2$	= general enthalpy functions, cal/mole
$\epsilon_1, \epsilon_2$	= enthalpy parameters for construction of general diagram
$\epsilon_1, \epsilon_2$	= enthalpy parameter functions of particular gas fulfilling general enthalpy equation
$\lambda, \lambda_0$	= constant ratio $\epsilon_2/\epsilon_1, \sigma_2/\sigma_1$ , respectively
$n$	= number of moles
$P$	= pressure, either kg/m <sup>2</sup> or lb/ft <sup>2</sup>
$P_1/P_2$	= pressure ratio of compression or expansion
$Q_p$	= molar heat increment at constant pressure, cal/mole
$Q_v$	= molar heat increment at constant volume, cal/mole
$R$	= universal gas constant, 1.98719 cal/mole-°K
$S$	= relative molar entropy at constant pressure = $\int_{298.16}^T \frac{C_p}{T} dT$ , cal/mole-°K

$S_v$	= relative molar entropy at constant volume = $\int_{298.16}^T \frac{C_v}{T} dT$ , cal/mole-°K
$\Delta S'$	= entropy increment for change in pressure, cal/mole-°K
$\Delta S'_v$	= entropy increment for change in volume, cal/mole-°K
$S'$	= total relative entropy for invariant composition = $S + \Delta S' = S_v + \Delta S'_v$ , cal/mole-°K
$s_0, s_1, s_2$	= general entropy functions, cal/mole-°K
$\sigma_1, \sigma_2$	= entropy parameters for construction of general diagram
$\sigma_1, \sigma_2$	= entropy parameter functions of particular gas fulfilling general entropy equation
$T$	= absolute temperature, °K
$T_0$	= reference temperature for zero values of enthalpy, internal energy and entropy, = 298.16°K
$V_1/V_2$	= volume ratio of compression or expansion

## Subscripts

0, 1, 2	= in connection with lower case letters: Relating to one of the three general functions
1, 2	= in connection with thermal functions and parameters: Relating to left-hand side (1), or to right-hand side (2) of diagram
I, II, III	= one of the three standard gases
$i$	= a particular gas
$m$	= average value for gas mixtures
$e$	= enthalpy
$p$	= pressure
$s$	= entropy
$v$	= volume

## Other Symbols

$\approx$	= value nearly constant over wide range of temperature
* and **	= assumed numerical values for standard gas parameters

## Introduction

THE easiest, most perspicuous, and consequently the most reliable method for thermodynamic calculations consists of using charts, such as T-S, T-H, H-S (Mollier), or P-H diagrams. However, this method has not yet generally been introduced in jet propulsion thermodynamics. The reason for this is that a particular diagram must be constructed for each individual working agent. This is a cumbersome process, and, consequently, diagrams are available for only a few of the most commonly used substances, such as steam, air, CO<sub>2</sub>, refrigerants, and for combustion products of particular propellant mixtures (1, 2).<sup>3</sup> The method fails completely for the majority of rocket applications where the working agent consists of a mixture of numerous gases, or when a multitude of different combinations has to be compared.

Consequently, attempts have been made in the past to develop general diagrams which can be used for a multitude of substances and their mixtures by attributing to them different, but constant, parameters. This proved to be possible if such diagrams were restricted to ideal gases [defined by

<sup>3</sup> Numbers in parentheses indicate References at end of paper.

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<sup>2</sup> In all units, cal/mole can be replaced by Kcal/Kmole.



$(\partial H/\partial P)_T = 0$ ] and to ideal gas mixtures of invariant composition. The first of these limitations is not particularly harmful for usual applications in the high temperature range where the derivation from perfect gas behavior is ordinarily negligible. The second limitation restricts use of such diagrams to "frozen composition" applications. This condition, however, is normally assumed to prevail in jet propulsion cycles due to their high time rate of expansion.

The important fact is that these charts can be made to incorporate the effects of temperature variation of the specific heats for a multitude of gases. This is achieved by assuming a general equation for the specific heat in which the particular gases are represented by appropriate coefficients or parameters.

Stodola (3) introduced, about 30 years ago, a linear general function for the molar specific heat,  $C_p = 6.67 + bT$ , where  $b$  was a constant parameter characterizing a particular gas. Gas mixtures were represented by an average parameter which was obtained as the weighted average of the parameters of the components. Based on this general function, he constructed a general temperature-entropy diagram over a relatively wide range of temperatures; this chart could be used as a Mollier chart for the usual combustion gases by introducing simple graphical corrections.

The linear approximation for the specific heat, however, is not sufficient for the temperature range and for the variety of gases encountered in jet propulsion cycles. Lutz (4) developed, some 15 years ago, a more advanced chart for the requirements of air compressor and gas turbine calculations. This diagram could also be used like a Mollier diagram for different gases and gas mixtures by applying graphical corrections. It was based on the assumed relationship

$$C_p = f_1(t) + \beta f_2(t) \dots \dots \dots [1]$$

where  $f_1$  and  $f_2$  are general, nonlinear functions of temperature only, and  $\beta$  is the constant parameter representing a particular gas. However, according to its purpose, this diagram was constructed only for temperatures up to 1400 C, and a sufficient accuracy was obtainable with constant parameters only for  $H_2O$ ,  $CO_2$ , air,  $O_2$ ,  $N_2$ , and for the mixtures thereof.

A general enthalpy and entropy diagram extending to 5000 K and permitting sufficiently accurate representation of at least all bi- and triatomic gases and their mixtures by constant parameters is described in this paper. It was made possible by adding a third term to the Lutz equation, by selecting appropriate general functions, and by introducing an optimization process in the determination of the parameters of individual gases.

## Derivation

A general equation can be assumed for the molar specific heat

$$C_p(T) = c_0(T) + \gamma_1 c_1(T) + \gamma_2 c_2(T) \dots \dots \dots [2]$$

which is suited for graphical representation in a two-dimensional diagram. In this equation,  $c_0$ ,  $c_1$ , and  $c_2$  are general functions of temperature only, and  $\gamma_1$  and  $\gamma_2$  are a pair of parameters identifying particular gases. In general, these parameters are also functions of temperature. For a practical general diagram, however, these parameters ought to be constant, or nearly constant, over a wide range of temperature. Therefore, specific general functions must be found which reduce the general equation to a form permitting the representation of the specific heats of individual gases (subscript  $i$ ) by nearly constant (symbol  $\approx$ ) parameters

$$C_{pi}(T) = c_0(T) + \tilde{\gamma}_{1i} c_1(T) + \tilde{\gamma}_{2i} c_2(T) \dots \dots \dots [3]$$

Plotting the molar specific heat of various gases against temperature, markedly similar curves are obtained above

room temperature for all bi- and triatomic gases. This similarity leads to a conclusion that specific general functions, Equation [3], might be obtained from the specific heats of a few representative gases by simple additive operations. Since the general equation is composed of three general functions, it is obvious that this system permits, and requires, the selection of three representative gases to which constant parameters can be deliberately attributed. These representative gases will henceforth be called "standard gases" and be denoted by the subscripts I, II, III.

The relative molar enthalpy of ideal gases

$$H(T) = \int_{298.16}^T C_p dT \dots \dots \dots [4]$$

is a function of temperature only; consequently, the equation for the general enthalpy-temperature diagram can be obtained by integration of Equation [2]

$$\begin{aligned} H(T) &= e_0(T) + \epsilon_1 e_1(T) + \epsilon_2 e_2(T) \} \\ &= H_1(T) + H_2(T) \} \dots \dots \dots [2h] \end{aligned}$$

where, analogously,  $e$  are general enthalpy functions of temperature only, and  $\epsilon_1$  and  $\epsilon_2$  are a pair of parameters identifying individual gases. Again, a practical diagram is obtained if it becomes possible to derive such general functions so that the molar enthalpy of individual gases,  $H_i(T)$  can be represented by the general Equation [2h] with constant, or nearly constant parameters  $\tilde{\epsilon}_{1i}$  and  $\tilde{\epsilon}_{2i}$

$$H_i(T) = e_0(T) + \tilde{\epsilon}_{1i} e_1(T) + \tilde{\epsilon}_{2i} e_2(T) \dots \dots \dots [3h]$$

The total molar entropy is not a function of temperature only. Restricting this method to individual gases and to gas mixtures of invariant composition, the total relative molar entropy  $S'$  can be written as

$$S' = S + \Delta S' \quad \text{and} \quad S'_v = S_v + \Delta S'_v \dots \dots \dots [5]$$

where  $S$  and  $S_v$  denote that portion of the relative molar entropy which is a function of temperature only

$$S = \int_{298.16}^T \frac{C_p}{T} dT \dots \dots \dots [5p]$$

or

$$= \int_{298.16}^T \frac{C_v}{T} dT \dots \dots \dots [5v]$$

and  $\Delta S'$  and  $\Delta S'_v$  denote the molar entropy increments resulting from changes in pressure or in specific volume, respectively

$$\Delta S' = R \ln P_1/P_2 \dots \dots \dots [6p]$$

$$\Delta S'_v = R \ln V_2/V_1 \dots \dots \dots [6v]$$

Since  $S$  is a function of temperature only, its general equation can also be obtained by integration of Equation [2], resulting in the equations

$$\begin{aligned} S(T) &= s_0(T) + \sigma_1 s_1(T) + \sigma_2 s_2(T) \} \\ &= S_1(T) + S_2(T) \} \dots \dots \dots [2s] \end{aligned}$$

for the construction of the general entropy-temperature diagram, and

$$S_i(T) = s_0(T) + \tilde{\sigma}_{1i} s_1(T) + \tilde{\sigma}_{2i} s_2(T) \dots \dots \dots [3s]$$

for the representation of the entropy of individual gases or mixtures.  $s(T)$  are general functions of temperature only, and  $\sigma$  are parameters identifying individual gases which, for practicability, must be reduced to near constant values  $\tilde{\sigma}$ .

Parameters of gas mixtures,  $\epsilon_m$  and  $\sigma_m$  are easily obtained as the weighed average value of the component parameters

$$\epsilon_m = \frac{\sum \epsilon_i n_i}{\sum n_i} \quad \text{and} \quad \sigma_m = \frac{\sum \sigma_i n_i}{\sum n_i} \dots \dots \dots [7h], [7s]$$

where  $n_i$  = mole number of individual components.

The reduction of the general equation to its specific form with nearly constant parameters for individual gases was made possible by (a) determining suitable general functions, (b) selecting appropriate standard gases, and (c) introducing an optimization process in the calculation of the individual gas parameters, as follows:

1 *Determination of the General Functions:* The general functions are determined by assigning constant, deliberately selected parameters  $\epsilon$  and  $\sigma$  to three standard gases for which the relative molar enthalpy  $H_I, H_{II}, H_{III}$ , and relative molar entropy at constant pressure  $S_I, S_{II}, S_{III}$  are known over the entire temperature range. This results in two sets of three simultaneous, linear equations

$$\begin{aligned} H_I(T) &= e_0(T) + \epsilon_{1I}e_1(T) + \epsilon_{2I}e_2(T) \\ H_{II}(T) &= e_0(T) + \epsilon_{1II}e_1(T) + \epsilon_{2II}e_2(T) \\ H_{III}(T) &= e_0(T) + \epsilon_{1III}e_1(T) + \epsilon_{2III}e_2(T) \end{aligned} \quad [6h]$$

and

$$\begin{aligned} S_I(T) &= s_0(T) + \sigma_{1I}S_1(T) + \sigma_{2I}S_2(T) \\ \dots &= \dots + \dots + \dots \\ \dots &= \dots + \dots + \dots \end{aligned} \quad [6s]$$

which can be solved for the general functions. It can readily be seen that these two functions are still fully determined if only two different numerical values for the standard gas parameters are used; for example, by assuming for each standard gas the same parameters for entropy and for enthalpy, and by specifying

$$\begin{aligned} \bar{\epsilon}_{1I} &= \bar{\epsilon}_{2I} = \bar{\epsilon}_{2II} = \bar{\epsilon}_{1III} = \bar{\epsilon}^* = \bar{\sigma}^* \\ \bar{\epsilon}_{1II} &= \bar{\epsilon}_{2III} = \bar{\epsilon}^{**} = \bar{\sigma}^{**} \end{aligned}$$

Substituting these values for the parameters into Equations [6h] and [6s], and solving for the general functions  $e(T)$  and  $s(T)$ , it becomes apparent that a further simplification is possible by assuming  $\bar{\epsilon}^{**} = \bar{\epsilon}^* + 1$ . Selecting, finally

$$\bar{\epsilon}^* = \bar{\sigma}^* = 1$$

and

$$\bar{\epsilon}^{**} = \bar{\sigma}^{**} = 2$$

the following formulas are obtained for the calculation of the general functions  $e(T)$  and  $s(T)$  from the known enthalpies and entropies of the three standard gases

$$\begin{aligned} e_0(T) &= 3H_I(T) - [H_{II}(T) + H_{III}(T)] \\ e_1(T) &= H_{II}(T) - H_I(T) \\ e_2(T) &= H_{III}(T) - H_I(T) \end{aligned} \quad [8e]$$

and

$$\begin{aligned} s_0(T) &= S_I(T) - [S_{II}(T) + S_{III}(T)] \\ s_1(T) &= S_{II}(T) - S_I(T) \\ s_2(T) &= S_{III}(T) - S_I(T) \end{aligned} \quad [8s]$$

2 *Selection of Standard Gases:* Proper selection of the standard gases was the most important means for obtaining nearly constant parameters for the other gases. It is a fortunate occurrence that the three most common gases,  $N_2$ ,  $H_2O$ , and  $CO_2$ , are also representatives for the three most different patterns of  $C_p$  curves vs. temperature. This made it possible to obtain the thermal functions of other gases with high accuracy by the introduction of constant parameters into the additive general equation. In addition, attribution of constant parameters to the most commonly encountered gases created the best chance for obtaining nearly constant parameters for usual gas mixtures. Selection of these three gases as standard gases, therefore, was a logical choice.

Finally, positive general functions were obtained by specifying  $N_2$  as gas I,  $H_2O$  as gas II, and  $CO_2$  as gas III. The general enthalpy and entropy functions obtained with three standard gases are presented in Table 1.

3 *Determination of Parameter Functions Approaching Constant Values for Other Than the Standard Gases:* The parameter functions  $\epsilon_1(T)$ ,  $\epsilon_2(T)$  and  $\sigma_1(T)$ ,  $\sigma_2(T)$  of individual gases are defined as the functions which fulfill the general

Table 1a General enthalpy functions

$T, ^\circ K$	$e_0$	$e_1$	$e_2$
200	-449.169	-101.210	-132.625
210	-397.925	-91.114	-124.375
220	-348.149	-80.909	-114.821
230	-299.404	-70.758	-104.127
240	-251.909	-60.552	-92.239
250	-205.767	-50.239	-79.158
260	-160.715	-39.925	-64.935
270	-116.908	-29.558	-49.519
280	-74.350	-19.136	-32.909
290	-32.935	-8.604	-15.267
298.16	0.000	0.000	0.000
300	7.235	2.035	3.514
310	46.373	12.728	23.327
320	84.316	23.530	44.226
330	121.065	34.441	66.210
340	156.675	45.514	89.171
350	191.251	56.697	113.110
360	224.905	67.933	137.972
370	257.421	79.387	163.755
380	288.795	91.003	190.517
390	319.413	102.727	218.037
400	348.888	114.615	246.535
450	483.668	176.226	400.480
500	598.639	241.690	572.282
550	697.810	310.629	759.122
600	783.631	383.041	958.934
650	858.922	458.601	1,169.76
700	925.470	537.202	1,390.15
750	984.692	618.951	1,618.73
800	1037.67	703.902	1,854.42
850	1085.05	792.274	2,096.30
900	1127.34	884.227	2,343.61
950	1164.85	979.981	2,595.70
1000	1197.85	1,079.69	2,851.96
1050	1226.52	1,183.59	3,111.87
1100	1250.72	1,291.72	3,375.19
1150	1270.59	1,404.30	3,641.50
1200	1286.55	1,521.11	3,910.41
1300	1305.60	1,768.37	4,455.29
1400	1308.37	2,033.32	5,008.27
1500	1295.89	2,315.37	5,567.75
1600	1268.53	2,613.92	6,132.88
1700	1227.28	2,923.11	6,702.63
1800	1172.29	3,257.00	7,277.04
1900	1104.81	3,600.23	7,854.60
2000	1027.03	3,955.07	8,435.31
2100	937.252	4,322.34	9,019.60
2200	837.914	4,700.69	9,605.85
2300	728.970	5,089.19	10,194.8
2400	611.394	5,487.18	10,785.7
2500	485.515	5,893.92	11,378.9
2600	352.262	6,308.90	11,973.5
2700	211.932	6,731.38	12,570.2
2800	65.208	7,160.92	13,168.1
2900	-88.193	7,597.02	13,768.0
3000	-247.348	8,039.20	14,369.3
3200	-581.670	8,940.45	15,575.9
3400	-935.643	9,861.56	16,788.1
3600	-1306.06	10,800.2	18,004.3
3800	-1692.55	11,754.4	19,225.7
4000	-2093.86	12,722.7	20,452.1
4200	-2057.22	13,703.2	21,682.2
4400	-2932.35	14,694.7	22,916.7
4600	-3368.46	15,696.3	24,155.5
4800	-3814.82	16,707.1	25,398.6
5000	-4270.03	17,725.5	26,646.1

Equations [2h] and [2s]. They are calculated from these equations by introducing the general entropy and enthalpy functions and the actual values of relative molar enthalpy  $H_i(T)$  and of relative molar entropy  $S_i(T)$  at constant pressure for

**Table 1b General entropy functions**  
cal(mole)<sup>-1</sup>(°K)<sup>-1</sup> or (Kcal(Kmole)<sup>-1</sup>(°K)<sup>-1</sup>

T, °K	s <sub>0</sub>	s <sub>1</sub>	s <sub>2</sub>
200	- 1.84296	-0.41130	- 0.52413
210	- 1.59357	-0.36182	- 0.48359
220	- 1.36088	-0.31472	- 0.43967
230	- 1.14466	-0.26942	- 0.49198
240	- 0.94277	-0.22590	- 0.34130
250	- 0.75438	-0.18377	- 0.28785
260	- 0.57792	-0.14323	- 0.23201
270	- 0.41238	-0.10428	- 0.17379
280	- 0.25778	-0.06613	- 0.11357
290	- 0.11253	-0.02916	- 0.05157
298.16	0.00000	0.00000	0.00000
300	0.02439	0.00661	0.01182
310	0.15257	0.04178	0.07680
320	0.27261	0.07615	0.14337
330	0.38586	0.10974	0.21094
340	0.49217	0.14273	0.27950
350	0.59273	0.17512	0.34865
360	0.68734	0.20691	0.41859
370	0.77615	0.23831	0.48954
380	0.86040	0.26912	0.56068
390	0.93932	0.29971	0.63241
400	1.01440	0.32973	0.70436
450	1.33199	0.47478	1.06681
500	1.57520	0.61250	1.42829
550	1.76419	0.74405	1.78419
600	1.91341	0.87004	2.13196
650	2.03424	0.99086	2.46938
700	2.13301	1.10731	2.79587
750	2.21468	1.22018	3.11124
800	2.28343	1.32968	3.41528
850	2.34086	1.43679	3.70859
900	2.38875	1.54211	3.99137
950	2.42929	1.64564	4.26401
1000	2.46327	1.74798	4.52672
1050	2.49049	1.84913	4.78128
1100	2.51375	1.94988	5.02530
1150	2.53066	2.04983	5.26296
1200	2.54415	2.14959	5.49150
1300	2.56047	2.34711	5.92708
1400	2.56285	2.54345	6.33644
1500	2.55489	2.73780	6.72216
1600	2.53741	2.93055	7.08661
1700	2.51178	3.12112	7.43238
1800	2.47877	3.30912	7.76226
1900	2.44361	3.49412	8.07385
2000	2.40386	3.67655	8.37113
2100	2.36034	3.85559	8.65610
2200	2.31346	4.03165	8.92933
2300	2.26517	4.20434	9.19104
2400	2.21549	4.37365	9.44222
2500	2.16322	4.53958	9.68546
2600	2.11115	4.70253	9.91816
2700	2.05868	4.86191	10.1429
2800	2.00464	5.01810	10.3611
2900	1.94998	5.17131	10.5721
3000	1.89713	5.32114	10.7750
3200	1.78922	5.61187	11.1645
3400	1.68251	5.89107	11.5313
3600	1.57561	6.15934	11.8799
3800	1.47287	6.41708	12.2087
4000	1.36913	6.66548	12.5239
4200	1.26740	6.90473	12.8248
4400	1.16903	7.13545	13.1113
4600	1.07283	7.35802	13.3860
4800	0.97745	7.57303	13.6511
5000	0.88485	7.78089	13.9054

the particular gas in question. Since two parameters are to be calculated from one equation, a degree of freedom remains which was used for a further optimization toward high accuracy with constant parameters. Two methods were applied successfully: either use of a fixed ratio between the two parameters  $\epsilon_2/\epsilon_1 = \lambda_{\epsilon}$  and  $\sigma_2/\sigma_1 = \lambda_{\sigma}$ , or use of a constant value for one of the parameters. Optimization was accomplished by calculating the parameters for different assumptions and finally selecting the one giving the smallest error in enthalpy and entropy for an average constant parameter value.

## Description and Use of Diagram

The large temperature range and the wide variety of gases for which this general diagram is intended made it impractical to develop an enthalpy-entropy (Mollier) diagram. Separate diagrams for enthalpy vs. temperature and for the entropy temperature function were constructed. However, these two diagrams are combined into a single system by plotting them against the same temperature abscissa, and simple graphical corrections are included so that internal energy and entropy at constant volume can be read directly from the chart. Finally, applying the general relationship for the molar entropy change with pressure, or with volume, Equations [6p] and [6v], these two systems can be used simultaneously to fulfill the functions of a true Mollier diagram for most practical purposes.

Use of the diagram is limited to ideal gases defined by  $(\partial H/\partial P)_T = 0$ , and to mixtures of ideal gases of invariant composition. The latter limitation requires, if dissociation has to be considered, that the composition of the gas be calculated by separate methods and that "frozen composition" be assumed within a cycle calculation. Apart from these two limitations, the values obtained from this diagram are not approximations but the true values for enthalpy and entropy including the effect of temperature variable specific heats of individual gases.

Two general diagrams were drawn in large scale (21 × 45 in.).<sup>4</sup> One covers the temperature range from 298.16 K (= 25 C = 536.69 R = 77 F) to 5000 K (= 9000 R) and is primarily intended for use in rocket cycle calculations. The other extends over the temperature range from 200 K (= -73.16 C = 360 R = -99.69 F) to 1700 K (= 1426.84 C = 3060 R = 2600.31 F) and is suited for calculations using free air and/or mechanical machinery (ramjets, ducted rockets, turbines, compressors, etc.).

Fig. 1 is a reduction of the large temperature range diagram. It contains two curve systems: one for the relative molar enthalpy, and one for the relative molar entropy at constant pressure, which were obtained from Equations [2h] and [2s], respectively, by introducing the general functions tabulated in Table 1 and various constant parameters. Values for intermediate parameters are obtained by linear interpolation between adjacent curves.

Fig. 2 shows, schematically, these two systems and how they are combined into a single enthalpy-temperature-entropy diagram by plotting them against a common temperature abscissa. Each system consists of two families of curves,  $H_1(T)$  and  $H_2(T)$  for enthalpy, and  $S_1(T)$  and  $S_2(T)$  for entropy. In both systems, the family designated by the subscript 1 is plotted to the left, and the family designated by the subscript 2 to the right side of the temperature abscissa. By this means, the value of a particular gas at a given temperature is directly obtained as the distance between two points on two curves which are defined by the corresponding two parameters of the gas in question,  $\epsilon_1, \epsilon_2$  for enthalpy, and  $\sigma_1, \sigma_2$  for entropy.

<sup>4</sup> Full-sized, unfolded, copies of the diagrams may be obtained on request to Commander, U. S. Naval Ordnance Test Station, Inyokern, Attn.: Code 7511, China Lake, Calif.



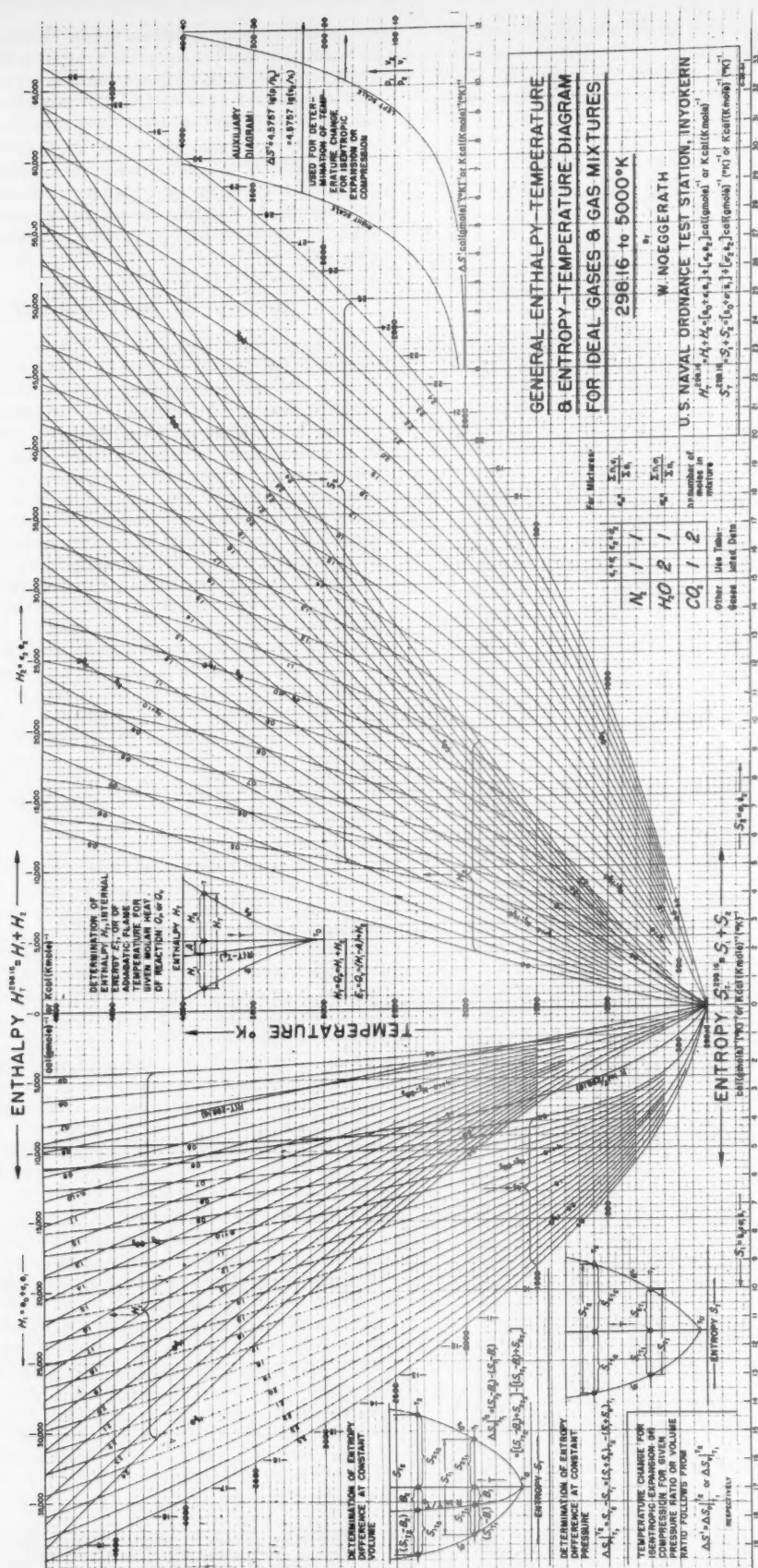


Fig. 1 General enthalpy-temperature-entropy diagram for ideal gases and gas mixtures 298.16° to 5000°K



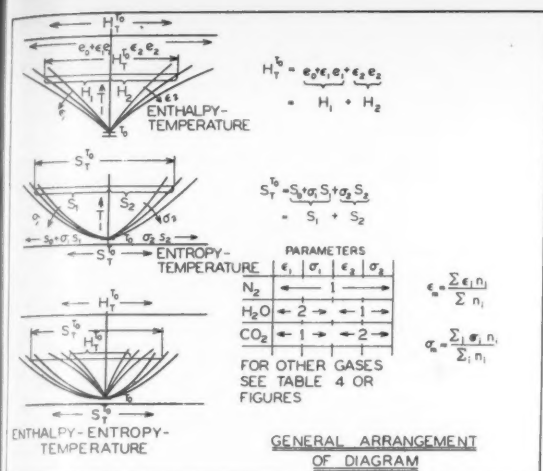


Fig. 2 General arrangement of diagram

Three auxiliary curves are added to the diagram for the graphical determination of the molar internal energy, of the entropy at constant volume, and of the temperature change at isentropic expansion or compression. The addition of these three curves makes the general diagram a full substitute for the Mollier chart. Their use will be explained below in discussing the basic operations for which they are intended.

The first step in using the diagram is the determination of the parameters for the gas, or gas mixture, for which the calculation is to be made. For the three most common gases, N<sub>2</sub>, H<sub>2</sub>O, and CO<sub>2</sub>, the parameters are constants and have the following simple numerical values

	$\epsilon_1 = \sigma_1$	$\epsilon_2 = \sigma_2$
N <sub>2</sub>	1	1
H <sub>2</sub> O	2	1
CO <sub>2</sub>	1	2

For other gases, the parameters are variable with temperature. They can easily be determined from their enthalpy and entropy data using Equations [2h] and [2s], respectively, in combination with one of the methods described under "3" in the foregoing section. The calculated parameters for the seven gases CO, H<sub>2</sub>, O<sub>2</sub>, OH, NO, O, and H are presented in Fig. 3<sup>5</sup> for the entire temperature range from 300 to 5000 K.

For gas mixtures, the parameters are calculated according to Equations [7] from the component parameters. Constant parameters can be used with adequate accuracy in almost all practical cases where gas mixtures contain a substantial amount of standard gases.

The most important basic operations which can be performed with the diagram are illustrated schematically in Figs. 4-7.

1 For a given temperature, the molar enthalpy is obtained as the distance between points in the left-hand and right-hand families of the  $H$ -curves as indicated by the particular values of the parameters  $\epsilon_1$  and  $\epsilon_2$  of the gas in question (Fig. 4, left side). The molar internal energy,  $E(T)$ , can also be obtained directly from the diagram by subtracting from the enthalpy the distance from the abscissa to the auxiliary line  $R(T - T_0)$  cal/mole (Fig. 4, right side).

2 The final temperature resulting from the addition or subtraction of heat to one mole of gas at either constant pressure,  $Q_p$ , or constant volume,  $Q_v$ , is determined by reversing the above operation for obtaining the enthalpy, or internal energy, for a given temperature. If a high degree of accu-

<sup>5</sup> Large-sized copies of gas parameter diagrams may be obtained on request to Cdr., U. S. Naval Ordnance Test Station, Inyokern, Attn.: Code 7511, China Lake, Calif.

## GAS PARAMETER FUNCTIONS

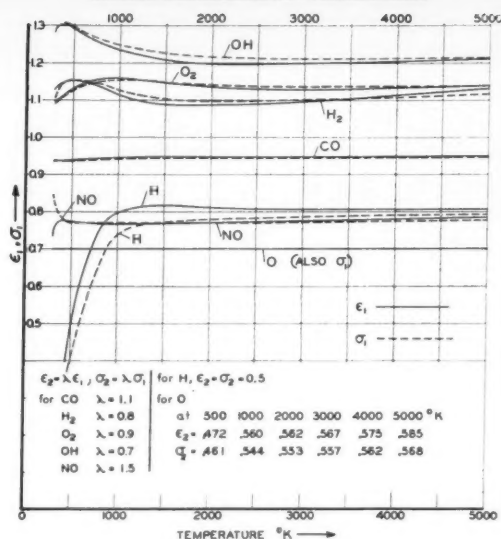


Fig. 3 Gas parameter functions

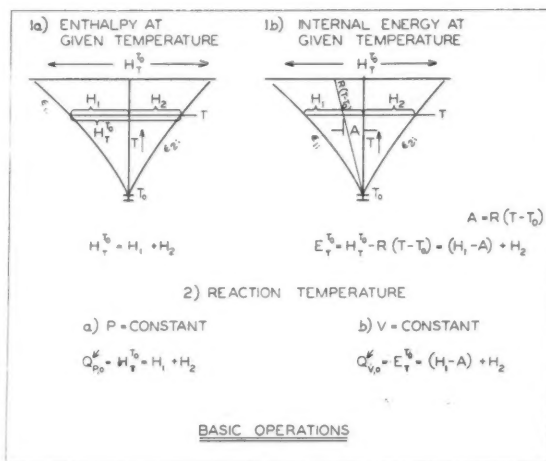


Fig. 4 Basic operations

racy is required, gas parameters  $\epsilon_1$  and  $\epsilon_2$  should be determined for a temperature close to the final temperature. This approximate final temperature can be determined from the diagram by inspection, using approximate component parameter values corresponding to the initial, or any intermediate temperature.

3 Determination of the molar entropy at constant pressure and at constant volume is accomplished from the system of entropy curves in an analogous way as was shown under "1" for the enthalpy and internal energy (Fig. 5). The auxiliary curve  $R \ln T/298.16$  cal/mole-°K is used for the graphical correction giving the entropy at constant volume.

4 The end temperature of an isentropic compression or expansion is found by determining the entropy increment,  $\Delta S'$ , corresponding to the compression or expansion ratio in question ( $\Delta S' = R \ln P_1/P_2$  if the pressure ratio is given, and  $\Delta S' = R \ln V_2/V_1$  if the volume ratio is given), and by adding or subtracting this increment to the entropy at the initial temperature (Fig. 6). The auxiliary diagram, on the right side of the general diagram, permits direct determination of the distance  $S_{T_0}^{T_1} - S_{T_2}^{T_1}$  for compression and expansion ratios up to 400:1. For most applications, use of parameters

selected for an approximate average temperature will give sufficiently accurate results. In order to obtain a very high degree of accuracy, the initial and final values of entropy should be determined for gas parameters corresponding to

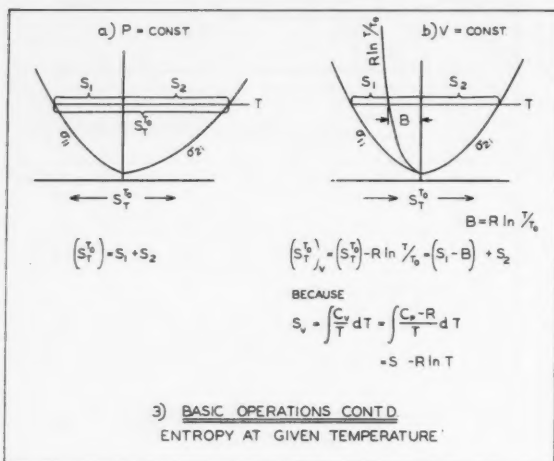


Fig. 5 Basic operations, continued

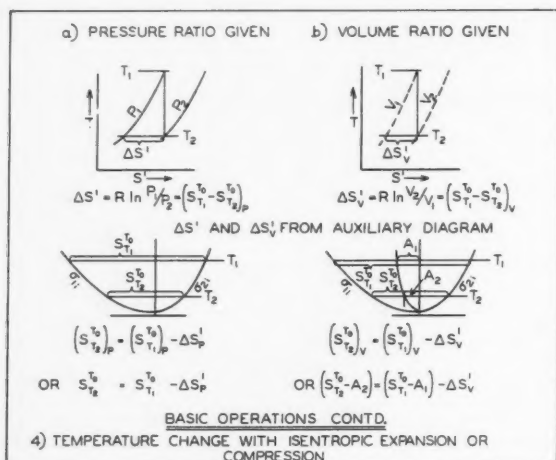


Fig. 6 Basic operations, continued

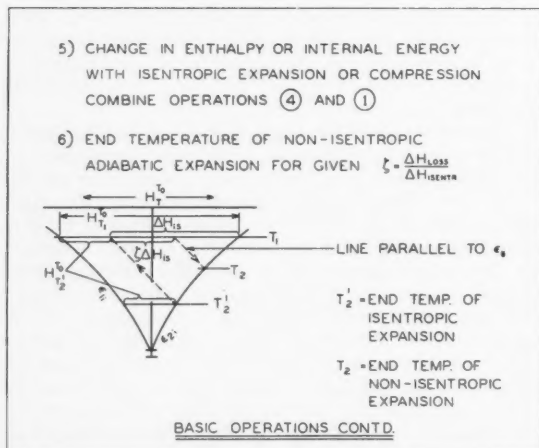


Fig. 7 Basic operations, continued

these temperatures. A trial-and-error method will lead to a quick solution.

5 The change in enthalpy or internal energy, for an isentropic compression or expansion at a given compression or expansion ratio, is the difference in enthalpies (pressure ratio given), or in internal energies (volume ratio given) for the initial and end temperatures of the isentropic compression or expansion. This is a combination of the afore-mentioned operations 1, 3, and 4.

6 The diagram can be used also for nonisentropic processes. Assuming, as an example, an adiabatic expansion with friction losses, the end temperature is found as that one for which the end-enthalpy  $H_2^T$  equals  $H_1^T + \Delta H_{\text{loss}}$  ( $H_1^T =$  end enthalpy for isentropic expansion). A simple graphical solution is shown schematically in Fig. 7.

The real value of this diagram, however, becomes apparent in the way more complex operations are simplified by its use.

As a first example, the determination of the specific impulse of a rocket propellant at high initial pressure and high expansion ratio is shown in Fig. 8. For this particular condition, the reaction gas can be assumed to be composed only of standard gases resulting in constant and simple gas parameters.<sup>4</sup> The entire complex calculation, consisting in the determination of the (fictitious) reaction temperature, initial enthalpy, initial entropy, entropy difference corresponding to given expansion ratio, end-entropy, end-temperature, end-enthalpy, enthalpy drop, exhaust velocity, and specific impulse, reduces to the drawing of two lines in the diagram and to a few elementary calculations which can be performed in a few minutes.

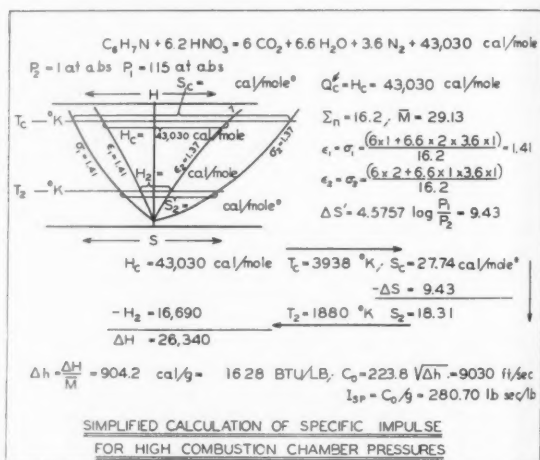


Fig. 8 Simplified calculation of specific impulse for high combustion chamber pressures

As a second example, the determination of a rocket nozzle configuration (exit area and throat area) for given initial conditions (pressure ratio and weight flow rate  $\dot{w}$ ) is demonstrated schematically in Fig. 9. As a first step (not shown in the figure), the expansion end-temperature is found according to the method described above (operation 4). Then a number of temperature intervals are assumed between the initial and the end-temperature. For each of these intermediate temperatures, the enthalpy and the entropy of the working gas are read from the diagram, and the corre-

<sup>4</sup> Specific impulses (equilibrium composition expansion) for high initial pressures and large expansion ratios can be determined with a high degree of accuracy by neglecting dissociation. This is because the high pressure reduces dissociation and because the large expansion ratio increases the amount of heat reintroduced into the system by recombination during the expansion. Note that the reaction temperature obtained by this method is too high and without any physical significance.

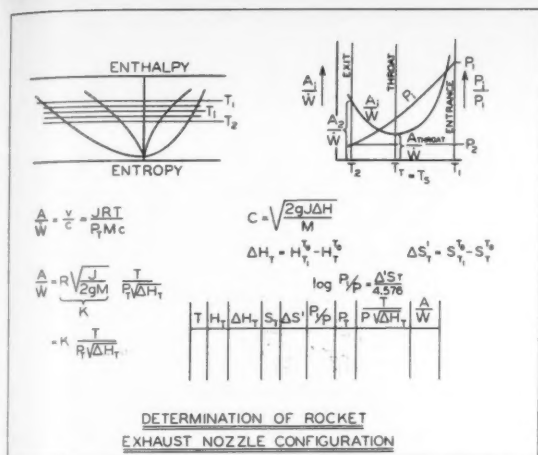


Fig. 9 Determination of rocket exhaust nozzle configuration

sponding values of pressure and of  $A/\dot{w}$  are calculated according to the derivation presented in Fig. 9 ( $A$  = variable nozzle area). Plotting the results vs. the expansion temperature, the exit area is obtained for the end-temperature,  $T_2$ , and the throat conditions (area, pressure, temperature) are obtained as those prevailing for the minimum value of  $A/\dot{w}$ .

It may be noted that, if the diagram is not available, or if an accuracy exceeding the limits of graphical representation is required, direct calculation of enthalpy and entropy from the general Equations [2h] and [2s] using the tabulated general functions, Table 1, is a very expeditious method with multicomponent gas mixtures. Compared to the classical method of calculating the thermal functions of a mixture at each temperature directly from the values of the components, the great advantage in using the general equations results from the fact that the cumbersome calculation of weighed average values has to be performed only once for the determination of the parameters of the mixtures.

### Discussion of Accuracy Obtained with Diagram Using Constant Gas Parameters

In order to take full advantage of the diagram, the parameters identifying individual gases are required to be nearly constant over a wide range of temperature. For the most common gases, N<sub>2</sub>, H<sub>2</sub>O, and CO<sub>2</sub>, these parameters are constants by the way this diagram was derived. For other gases, it was tried by the means discussed above to obtain nearly constant parameters. As seen in Fig. 3, this goal was remarkably well approached. Except for H<sub>2</sub>, the deviation from an average constant parameter does not exceed 1 per cent for temperatures between 1500 and 5000 K. Lowering the temperature to 500 K, major deviations are obtained only for gases which are usually not present in the low temperatures region (OH and monoatomic oxygen and hydrogen). For molecular hydrogen, the maximum deviation from a constant average value is 2.5 per cent for temperatures between 800 and 5000 K.

Gas mixtures encountered in jet propulsion cycle calculations usually contain a considerable amount of the standard gases N<sub>2</sub>, H<sub>2</sub>O, and CO<sub>2</sub>. It is obvious that the presence of standard gases will reduce the variation of parameters for the mixture. For example, 40 per cent of molecular hydrogen is required in order to effect a 1 per cent variation of the parameters of such mixtures.

The variation in enthalpy and entropy with temperature is even smaller than that of the corresponding parameters. This is due to the first terms in the general equations which are independent of the parameters.

In actual use of the diagram only very few occasions were encountered where constant parameters did not provide sufficient accuracy. The following examples are presented as evidence of the accuracy obtainable with constant parameters.

### 1 Enthalpy and Entropy of Two Rocket Reaction Cases

Two rocket reaction gases were selected for this investigation which can be considered as typical representatives for compositions giving high reaction temperatures (mixture "X") and low reaction temperatures (mixture "Y"):

Moles	Mixture "X"	Mixture "Y"
H <sub>2</sub> O	0.2621	0.2877
CO <sub>2</sub>	0.1715	0.0457
CO	0.3340	0.2877
H <sub>2</sub>	0.0638	0.3789
N <sub>2</sub>	0.1386	
O <sub>2</sub>	0.0022	
OH	0.0149	
NO	0.0027	
H	0.0091	
O	0.0013	

The calculated gas parameters for both mixtures are shown in Table 2 for a wide range of temperatures. For the high reaction temperature mixture "X," the deviation of the parameters from average values is so small that use of average constant parameters appears justified without further discussion. In addition, identical parameters can be used for enthalpy and entropy.

For the lower reaction temperature mixture "Y," however, the deviation from average parameters may be large enough to necessitate further investigation. Consequently, the mo-

Table 2 Gas Parameters for Two Rocket Reaction Gas Mixtures

Mixture "X" (high reaction temperature)

T, °K	$\epsilon_1$	$\epsilon_2$	$\sigma_1$	$\sigma_2$
3200	1.2518	1.1711	1.2515	1.1707
2800	1.2513	1.1708	1.2511	1.1708
2400	1.2508	1.1704	1.2510	1.1707
2000	1.2505	1.1702	1.2508	1.1705
1800	1.2505	1.1700	1.2509	1.1705
1600	1.2506	1.1701	1.2509	1.1705
1400	1.2509	1.1703	1.2512	1.1707
1200	1.2513	1.1706	1.2521	1.1710
1000	1.2521	1.1713	1.2518	1.1713
900	1.2524	1.1716	1.2519	1.1715
800	1.2525	1.1718	1.2520	1.1718
700	1.2525	1.1720	1.2519	1.1719
600	1.2519	1.1720	1.2512	1.1717
500	1.2506	1.1716	1.2501	1.1712
Range <sup>a</sup>	0.0020	0.0020	0.0020	0.0014
Relative range, per cent <sup>b</sup>	0.16	0.18	0.16	0.12

Mixture "Y" (low reaction temperature)

T, °K	$\epsilon_1$	$\epsilon_2$	$\sigma_1$	$\sigma_2$
2000	1.3057	1.0081	1.3094	1.0115
1800	1.3051	1.0081	1.3098	1.0117
1600	1.3057	1.0085	1.3106	1.0123
1400	1.3083	1.0097	1.3121	1.0135
1200	1.3103	1.0122	1.3145	1.0153
1000	1.3151	1.0159	1.3181	1.0184
900	1.3193	1.0183	1.3204	1.0199
800	1.3214	1.0209	1.3230	1.0220
700	1.3247	1.0234	1.3253	1.0238
600	1.3274	1.0253	1.3273	1.0253
Range <sup>a</sup>	0.0223	0.0172	0.0179	0.0138
Relative range, per cent <sup>b</sup>	1.71	1.71	1.37	1.36

<sup>a</sup> Difference between maximum and minimum value.

<sup>b</sup> Range divided by smallest absolute value.

Table 3 Errors in Enthalpy and Entropy Caused by Use of Constant Gas Parameters

Low reaction temperature rocket gas

Enthalpy $T_i$ , °K	Absolute error, cal/mole			Relative error, %		
	(a)	(b)	(c)	(a)	(b)	(c)
2000	0.00	+52.78	+115.66	+0.00	+0.36	+0.80
1800	+1.95	+46.78	+100.05	-0.02	+0.37	+0.80
1600	-15.50	+21.66	+65.73	-0.14	+0.20	+0.61
1400	-13.29	+16.59	+51.94	-0.15	+0.17	+0.37
1200	-22.44	$\pm 0.00$	+27.17	-0.31	$\pm 0.00$	+0.38
1000	-32.40	-15.74	+3.90	-0.59	-0.29	+0.07
900	-35.93	-22.23	-6.13	-0.77	-0.48	-0.13
800	-34.79	-23.95	-11.80	-0.90	-0.62	-0.31
700	-31.81	-23.25	-13.66	-1.04	-0.76	-0.45
600	-24.80	-19.11	-12.39	-1.09	-0.84	-0.54

Entropy $T_i$ , °K	Absolute error, cal/mole -°K			Relative error, %		
	(a)	(b)	(c)	(a)	(b)	(c)
2000	0.000	+0.051	+0.091	0.00	+0.33	+0.58
1800	-0.002	+0.044	+0.081	-0.01	+0.30	+0.55
1600	-0.010	+0.032	+0.066	-0.07	+0.24	+0.49
1400	-0.020	+0.018	+0.047	-0.16	+0.15	+0.38
1200	-0.041	$\pm 0.000$	-0.026	-0.37	$\pm 0.00$	+0.24
1000	-0.046	-0.020	+0.001	-0.49	-0.21	+0.01
900	-0.051	-0.028	-0.009	-0.60	-0.33	-0.11
800	-0.054	-0.034	-0.019	-0.72	-0.45	-0.25
700	-0.052	-0.035	-0.023	-0.80	-0.54	-0.36
600	-0.046	-0.032	-0.023	-0.88	-0.61	-0.43

Data calculated from general functions Table 1, using constant parameters: (a) = parameters at 2000 K; (b) = parameters at intermediate temperature (1200 K); (c) = arithmetic average value of variable parameters over entire temperature range.

lar enthalpy and entropy were calculated for the temperature range between 600 and 2000 K from the tabulated general functions (Table 1), for the following cases:

(1) Variable parameters (Table 2), giving the correct enthalpy and entropy values.

(2) Constant parameters, determined as (a) the parameters at the highest temperature; (b) the parameters at approximate intermediate temperature of 1200 K; and (c) the arithmetic average of the correct parameters within the temperature range.

Table 3 contains the absolute relative errors resulting from constant parameters. It appears that even with the parameters determined for the highest temperature the relative error does not become larger than about one per cent within the entire temperature range. Use of intermediate or average parameters reduces the error below the one per cent limit with no significant difference in the accuracy obtained with these two methods. Consequently, it appears feasible to use constant parameters simply determined as the parameters at an approximately average temperature within the temperature range considered.

## 2 Enthalpy Drop for Isentropic Expansion

This investigation was made only for the low reaction temperature rocket gas mixture "Y" of the foregoing example because this mixture gives the larger change in parameters with temperature. Assuming an initial temperature of 2000 K and an expansion pressure ratio of 50:1, the initial enthalpy and, entropy, the expansion end-values of entropy, temperature, and enthalpy, and the resulting enthalpy drop were determined for the following cases:

(1) Correct values: Use of variable parameters (Table 2) and calculation from the general functions (Table 1), giving the correct values.

(2) Approximate values: Use of constant parameters (parameters at 2000 K) and (a) calculation using the general functions (Table 1); (b) graphical determination from diagram.

The results are shown in Table 4. It follows for the calculated values that the relative error introduced by use of constant parameters is only 0.6 per cent for the expansion end-temperature and less than 0.1 per cent for the enthalpy drop. The values determined from the diagram show an equal error

Table 4 Isentropic Expansion  
Accuracy Obtained with Constant Parameters

	1. Correct data	2. Approximate data obtained with constant parameters	
		(a) Calculated from General Functions Table 1	(b) Read from General Diagram
Initial enthalpy, cal/mole	14,695	14,695	14,690
Initial entropy, cal/mole-°K	15.685	15.685	15.69
End-entropy, cal/mole-°K	7.911	7.911	7.92
End-temperature, °K	837.8	843.2	843
End-enthalpy, cal/mole	4,169	4,178	4,180
Enthalpy drop, cal/mole	10,526	10,518	10,510

Constant parameters = parameters at 2000°K; Low reaction-temperature rocket gas; initial temperature = 2000°K; Expansion pressure ratio = 50:1.

of 0.6 per cent for the expansion end-temperature, and an error of 0.15 per cent for the enthalpy drop.

In both examples the errors obtained with constant parameters were smaller than usually considered permissible in technical calculations.

## Data Sources

The original data for enthalpy and entropy were taken from Refs. (5-9). The relative enthalpies and entropies, with zero value at 25°C = 298.16 K, which were used for the construction of the diagram were obtained from the values of the references by subtracting the respective values at 298.16 K. In case the values of enthalpy and entropy at 298.16 K were not listed in the references, they were calculated from the  $C_p$  data listed in the references for 290 and 300 K by assuming a linear change of  $C_p$  with temperature within this interval.

The same method of interpolation was used for obtaining  
(Continued on page 470)



# A Versatile Ignition Delay Tester for Self-Igniting Rocket Propellants

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Ignition delay is an important criterion by which self-igniting liquid rocket propellants are evaluated. An ignition delay tester which combines simplicity of construction and operation with a high degree of flexibility has been developed. Important variables, such as propellant temperature, propellant ratio, and degree of mixing, can be controlled and reproduced accurately. Screening of additives and determination of optimum fuel blends are among the uses to which ignition delay studies may be applied. The tester is a valuable tool in propellant research.

## Introduction

SELF-IGNITION of liquid rocket propellants upon contact of fuel and oxidizer is frequently a desirable characteristic because it eliminates the need for complex mechanical and electrical starting devices in rocket engines. However, it is usually not enough that propellants be self-igniting. They should also have the quality of igniting very quickly after they are combined, i.e., have very short ignition delays. This is necessary in order to prevent accumulation of dangerous quantities of unburned propellants in the combustion chamber. Experience has shown that, in general, propellants which have short ignition delays give smooth starts and stable combustion in a rocket engine. Thus, ignition delay is an important criterion for predicting the starting and combustion characteristics of a liquid propellant pair; and a suitable means for measuring ignition delay is a powerful tool for conducting research on self-igniting propellants.

The requirements of an ignition delay tester for research purposes are the following:

- 1 The reproducibility of results should be good, in order that minimum number of determinations suffice to fix the delay value for a given set of conditions.

- 2 As many pertinent variables as possible should be controllable. Some of these variables are propellant temperature, propellant ratio, and degree of mixing. Also, the tester should be applicable to a large variety of propellants and not be restricted to any particular class of oxidizer or fuel.

- 3 The construction of the tester should be as simple and trouble-free as possible so that it is easy to operate. The unit should be compact enough to be used in a fume hood so that noxious and toxic products of combustion will be carried away safely.

- 4 Quantities of propellants used for each determination should be as small as possible, both for safety and to allow extensive research on scarce or costly materials.

## Description of Tester

An ignition delay tester has been developed which meets these requirements to a satisfactory degree. Fig. 1 is a dia-

gram showing a cross-sectional view of the unit assembled. The dimensions in the drawing are purposely distorted in order to show the essential features more clearly.

The tester is made up of four main parts, the first of which is the outer or oxidizer cylinder made of stainless steel. The interior of the cylinder is hollow and has a conical bottom. Near the base of the inverted cone is an exposed electrode insulated from the walls of the instrument by a Teflon insulator. The electrode passes through the wall and is brought out in the metal tube to a cable connector.

The inner or fuel cylinder is also constructed of stainless steel. The lower section is a hollow cone, the outer part fitting snugly into the conical bottom of the oxidizer cylinder. There are four holes drilled through the inverted base of the cone, spaced at 90-deg intervals around the periphery. These holes are directed downward at an angle toward the inner apex of the cone.

The third major part is the weight assembly. The weight itself is annular in shape and is made of brass. It is mounted on top of the fuel cylinder and can be fastened thereto by means of a winged set screw. A stainless steel bar is attached through Teflon insulators to the top of the weight. Through a hole in the center of the bar is fastened a stainless steel tubular electrode, the position of which is adjustable by means of a set screw. An electrical connector is mounted on the side of the weight, and there is a jumper connected between this and the bar which holds the electrode.

The fourth part is a trigger block made of Bakelite.

The tester is used in conjunction with a suitable electronic timer. Any instrument capable of measuring accurately in

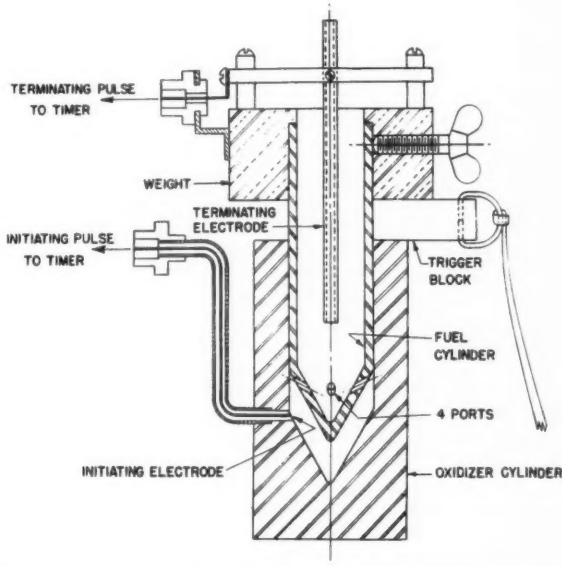


Fig. 1 Cross-sectional diagram for ignition delay tester

Presented at the ARS National Convention, New York, N. Y., December 1, 1954.

<sup>1</sup> Research Engineer.

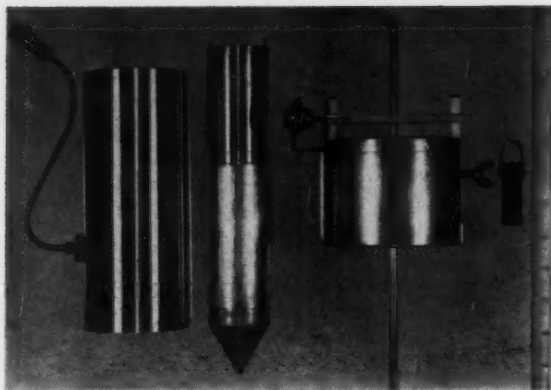


Fig. 2 Main parts of ignition delay tester

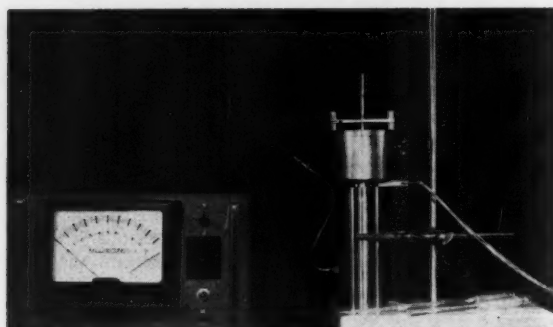


Fig. 3 View of assembled ignition delay tester

milliseconds the interval of time between two electrical pulses may be used.

Fig. 2 is a photograph of the disassembled parts of the tester. From left to right they are the oxidizer cylinder, the fuel cylinder, the weight assembly with the tubular electrode running through it, and the Bakelite trigger block. The scale gives an idea of the size of the tester, and it can be seen that its compactness allows it to be used in a fume hood with ease.

Fig. 3 shows the tester assembled and ready to be fired. The electronic timer in the photograph is a specially designed one, but there are electronic timers on the market which can be used in its place. The syringes shown lying on the base of the stand are used to measure the fuel and oxidizer.

### Operation

The operating sequence is as follows: The oxidizer cylinder is clamped in an upright position. By means of a modified hypodermic syringe having a six-inch tip made of 1-mm ID glass capillary tubing, an accurate volume, usually 3 ml, of oxidizer is introduced slowly at the apex of the conical bottom of the oxidizer cylinder. The weight assembly and the fuel cylinder are fastened together; the fuel cylinder is inserted into the oxidizer cylinder, the Bakelite trigger block being placed between the weight and the upper rim of the oxidizer cylinder. An accurate volume, usually 1 ml, of fuel is measured into the apex of the inner cone from a second syringe having a 9 or 10-inch tip made of 1-mm ID glass capillary tubing. The electrical connectors are connected to the electronic timer by means of shielded cables.

The unit is then ready to fire. When the trigger block is pulled out, the fuel cylinder descends under the influence of gravity, displacing the oxidizer upward and through the holes in the inner cylinder onto the surface of the fuel. As the oxidizer sweeps across the electrode in the outer cone, the electrode is grounded to the walls of the unit. This creates

the initiating electrical pulse for the electronic timer, and the measurement of the delay begins. Soon after the oxidizer contacts the fuel, ignition takes place. The ionization of the flame lowers the electrical impedance between the terminating electrode and the walls of the inner cylinder. This creates the terminating pulse for the electronic timer and completes the delay measurement.

The delay thus measured should be corrected for the interval of time required for the oxidizer to travel from the electrode in the outer cone to the surface of the fuel. This instrument factor is determined as follows: A small Teflon plug is inserted into the bottom end of the tubular electrode with the plug protruding slightly from the end. By means of the set screw, the position of the electrode is then adjusted so that the Teflon plug just touches the bottom of the fuel cylinder. The plug thus serves to position the electrode and to insulate it from the cylinder. The instrument is then operated under the desired conditions but without the fuel. The oxidizer sweeping across the initiating electrode transmits a pulse to the timer as in an ordinary run and furnishes the terminating pulse when it bridges the gap between the tubular electrode and the inner cone. The value of the instrument factor thus determined is usually between 6 and 10 ms, depending upon the individual tester. For any one instrument and for any given set of conditions, the instrument factor is remarkably constant and is reproducible within 1 ms.

After a run is completed, the weight assembly and the inner cylinder are cleaned with water and acetone and then dried. If another run is to be made with the same oxidizer, the residual oxidizer in the outer cylinder is drawn out with a pipette connected to a vacuum line. The tester is then ready to be reloaded. If a different oxidizer is to be used in the next run, the oxidizer cylinder is washed and dried before reloading.

With careful operation, the repeatability of results obtained with this ignition delay tester is very satisfactory. The repeatability, as a rule, depends upon the length of the ignition delay; and individual values are usually within 10 per cent of the average value in the range of 0 to 50 ms. Values over 50 ms are frequently considerably less repeatable.

Table 1 shows typical results obtained with the tester. For security reasons, the fuels are not identified. For the better fuels, as exemplified by A and B, the repeatability is such that it is seldom necessary to make more than three determinations at a given set of conditions. Fuel C is an example of a fuel that gives unusually poor repeatability, and with such materials it is best to make five or six determinations in order to obtain a satisfactory average value.

Table 1 Repeatability of ignition delay values

OXIDIZER = RFNA*		TEMPERATURE=75°F.
FUEL	CORRECTED IGNITION DELAY(MS)	AVERAGE IGNITION DELAY(MS)
A	8	9
	9	
	9	
B	22	22
	23	
	22	
C	132	99
	72	
	92	

\* RED FUMING NITRIC ACID

When the tester is to be used at temperatures other than room temperature, the procedure is modified only slightly. The tester is simply immersed to a depth of about 5 inches in a bath maintained at the desired temperature. The instrument temperature is obtained by means of a potentiometer box, the thermocouple of which is run through the tubular

electrode into the fuel. When the fuel reaches temperature equilibrium with the bath, the thermocouple is withdrawn and the tester operated as before. The tester has been operated satisfactorily at temperatures, ranging from -90 to +120 F, the only temperature limitations apparently being the freezing points of the propellants on the one hand and their boiling points on the other.

The number of runs that can be made in a given time with the tester depends on the temperature of operation. At room temperature, 10 or 12 runs can be made in an hour. At -40 F, about four runs per hour can be made. The difference is due to the additional time required to reach temperature equilibrium when operating at other than room temperature.

### Operating Variables and Typical Studies

A study of the effect of temperature on the ignition delays of two different fuels with the same oxidizer and using the same degree of mixing is shown in Fig. 4. This illustrates the importance of being able to vary and control the propellant temperature in ignition delay studies. Although, at room temperature, both fuels have short ignition delays, at low temperatures one of them has unsatisfactorily long delays.

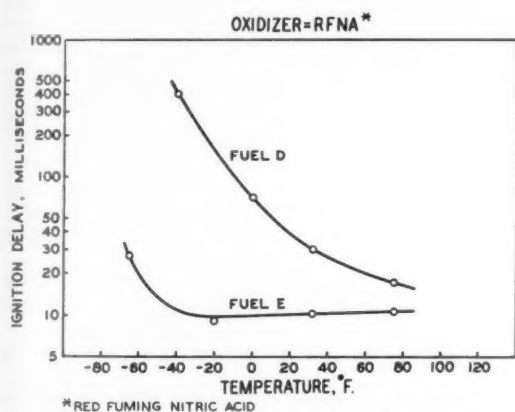


Fig. 4 Effect of temperature on ignition delay of two fuels

Propellant ratio can be varied with the tester by changing the volume of oxidizer and/or fuel loaded into the tester. However, most studies have shown that few propellant combinations are sensitive to propellant ratio in so far as ignition delay is concerned. Consequently, the propellant ratio has been standardized at 3 ml oxidizer to 1 ml fuel; and deviation from this ratio has seldom been necessary.

The small quantities of propellants required for an ignition delay determination make it possible to screen scarce or costly research samples. Also, these small test quantities are advantageous in that some propellants ignite with explosive violence, and ignition of larger quantities could be hazardous.

The tester has been used with various oxidizers and many fuels successfully.

One of the important variables which can be controlled in the tester is degree of mixing of the propellants. Many oxidizer and fuel combinations are sensitive to this variable; and, because most rocket engines produce less than optimum mixing, it is desirable to know the sensitivity to mixing of a given combination. The degree of mixing in the tester is varied simply by using differently sized trigger blocks. This varies the height that the fuel cylinder drops in operation; and this, in turn, varies the velocity with which the oxidizer is injected into the fuel. In Fig. 5, ignition delays of two fuels at two different temperatures using red fuming nitric acid (RFNA) as the oxidizer are plotted against height of drop of the fuel

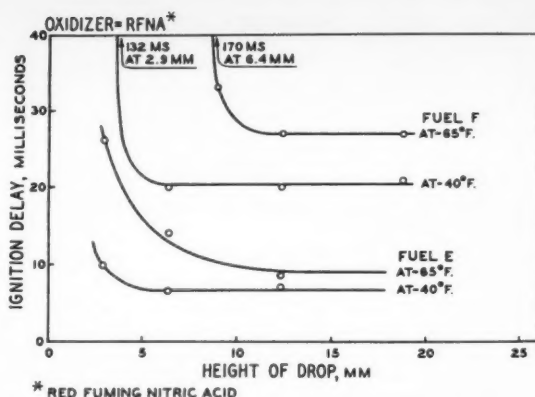


Fig. 5 Effect of degree of mixing on ignition delay

cylinder. The ignition delays become shorter as the drop heights increase, but this effect levels off at the greater drop heights. This leveling off has been found to occur with most propellant combinations. The delay value approached asymptotically by the level portion of the curve is believed to be the minimum delay that can be obtained through improvement of degree of mixing. This suggests that ignition delay values on the horizontal portion of the curve are almost entirely attributable to the chemical nature of the propellants, with mixing time contributing negligibly to this delay.

By determining the ignition delay of a propellant pair under conditions of both good and poor mixing, it should be possible to obtain an indication both as to the purely chemical ignitibility of the propellants and the ease with which the propellants can be adapted to the mixing conditions which exist in a rocket engine.

Fuel F in Fig. 5 may be used to illustrate this point. The flat portion of the curve at -65 F indicates a delay of about 27 ms with good mixing. This delay is short enough so that satisfactory ignition might be expected with this fuel in a rocket engine, but only if nearly optimum mixing is provided in the engine. With less than optimum mixing, the ignition delay rises rapidly and explosive starts could well result.

A number of other types of studies can be carried out with an ignition delay tester which yields dependable results. Frequently, the cost of a fuel that gives a short delay in combination with an oxidizer is greater than is desired. Dilution of the fuel with a cheaper, although perhaps less ignitable, fuel is a possible means of increasing availability and lowering the over-all fuel cost. In such a case, it is necessary to find the extent of dilution which can be tolerated before exceeding a limiting delay value. Fig. 6 shows the results of such a dilu-

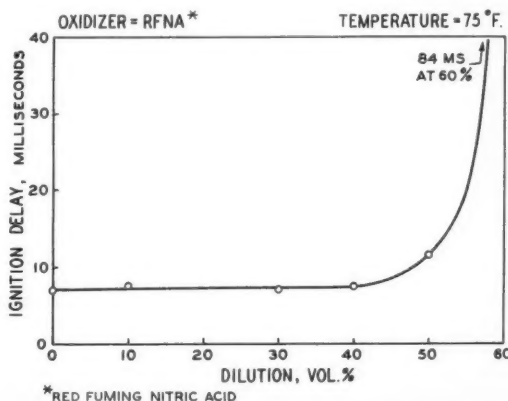


Fig. 6 Effect of dilution with jet fuel on ignition delay of fuel G

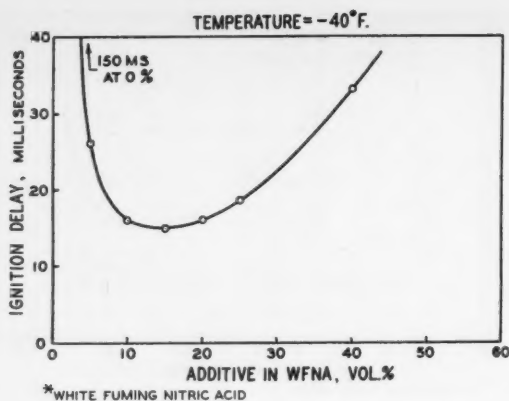


Fig. 7 Effect of additive content on activity of WFNA\*

tion study carried out at 75 F. The diluent, jet fuel, was not self-igniting with RFNA, the oxidizer used in this study. It can be seen that with 50 per cent dilution, the ignition delay is only 11.5 ms; and, consequently, this degree of dilution can probably be tolerated without incurring starting troubles in a rocket engine when operating at room temperature.

Fig. 7 presents one more type of study for which the ignition delay tester is useful. The number of oxidizers available for use in self-igniting systems is much smaller than the number of fuels. To some extent, the limited selection of oxidizers can be offset by the use of additives. Almost as important as deciding which additive to use is the question of how much additive to use. When the additive is for the purpose of improving ignitability, a study such as the one shown in Fig. 7 will determine the best additive content. In this case, about 15 per cent of the additive in white fuming nitric acid (WFNA) gives the shortest ignition delay with the fuel under consideration.

### Comparison with Other Ignition Delay Testers

There are in existence several other types of ignition delay testers ranging in complexity from simple drop testers to miniature rocket engines having elaborate instrumentation. Few references to specific test units can be made here because descriptions of them are for the most part in classified literature. However, almost all ignition delay testers fall into one of the following general classes. A comparison of the characteristics of each class with those of the tester described in this paper is presented.

#### Drop Testers

In this class of tester, a drop of one propellant is allowed to fall into a small vessel containing a small quantity of the other propellant (1).<sup>2</sup> Simple timing devices are usually used. Poor mixing is inherent in this class of tester. Control of propellant temperature is difficult. Even when control of the temperature of both fuel and oxidizer is provided, it is usually necessary for the falling drop to traverse a region the temperature of which is not at test temperature. Reproducibility of results is usually relatively poor. The quantities of propellants used in drop testers are, of course, smaller than those used in the subject tester.

#### Open-Cup Testers

These testers are similar to the drop tester except in scale. Usually 10 cc or more of one propellant is dropped into a vessel containing 10 cc or more of the other propellant. As in the drop tester, temperature control is awkward (2). Mixing, although better than in the drop tester, is almost

<sup>2</sup> Numbers in parentheses indicate References at end of paper.

never adequate nor can the degree of mixing be controlled. Reproducibility of results is seldom better than that obtained with the drop tester, and the quantities of propellants used per run are considerably greater than those used in the subject tester.

#### Impinging-Jet Testers

Testers of this class eject a stream each of fuel and oxidizer (3). The streams are impinged upon one another, and the interval between the instant of impingement and the instant of ignition is timed by various means (3). The degree of mixing obtained in impinging jets is ordinarily poor unless a splash plate is used. Temperature control is difficult. Reproducibility of results is fair in testers that employ a splash plate and poor in others. An advantage of this type of tester is that it can hold reservoirs of both propellants, and successive runs with the same propellants can be made quite rapidly. Approximately 1 cc of each propellant is used per run. The construction of impinging-jet testers is usually more complex than that of any of the above testers.

#### Miniature Rocket Engine Testers

These exist in many varieties and sizes. All are relatively complex in construction and instrumentation. Degree of mixing is dependent on the type of injector used. Repeatability of results can be good. Quantity of propellants used per run is usually over 200 cc, and the time between runs is frequently lengthy.

### Conclusions

The ignition delay tester described in this paper is a valuable instrument in rocket propellant research. It possesses the following important advantages:

- 1 It is simple and compact in construction and, therefore, easy to use in the laboratory.
- 2 The repeatability of results is good, obviating the need for a large number of runs to obtain a good average delay value.
- 3 The flexibility of operation allows study of a number of important variables.
- 4 The small quantities of propellants required per run make it a valuable tool for screening research samples for possible use as propellants.
- 5 The subject ignition delay tester combines the desirable features of other types to a considerable extent and also provides a few exclusive features.

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### Rocket Powered Test Vehicles

(Continued from page 445)

containers at the aft end of the missile. The final recovery parachute was mounted in the forward cabin section. A steel landing spike was provided.

The missile was designed to be carried aloft in the bomb bay of a modified B-29 aircraft (see Fig. 18). It is attached to the aircraft with a conventional 30-in. bomb rack.

In the Mach 1.5 test conducted on the Aberdeen Bombing





Fig. 18 Cherokee mounted in B-29 bomb bay

Mission Range at Edwards Air Force Base, the missile was released from the B-29 at an altitude of 32,000 ft. Immediately after the missile was released from the aircraft, a 24-ft Ribbon parachute was deployed. The purpose of this parachute was to orient the missile into a vertical trajectory before the JATO burning period to insure that the missile remained over the test range. When the missile was in a nearly vertical attitude, the stabilizing parachute was cut off, and the missile was allowed to free fall until it attained a speed of 500 mph. This occurred at an altitude of 25,000 ft. At this time the JATO unit was ignited and the missile accelerated to its maximum speed of Mach 1.5, at which time the escape device being tested was ejected. Cameras on the missile recorded the path of the seat as it left the missile. The seat itself is equipped with three mutually perpendicular rate gyros, three linear accelerometers, various air pressure gages, and a magnetic tape recorder. The method of mounting these instruments in the seat is shown in Fig. 19. In addition, ground-based phototheodolite cameras provide an accurate record of

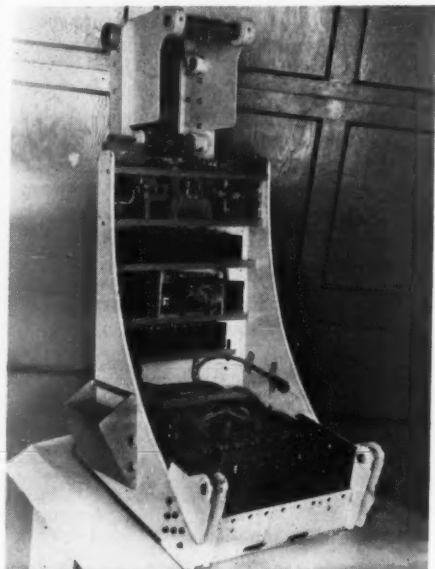


Fig. 19 Test seat with instrumentation

the entire operation. After ejection, the high drag of the missile caused it to decelerate to a speed of Mach 1.2. A parachute recovery system similar to that described for the Skokie missile was then set into operation. All operations were time sequenced.

Two tests have been conducted with this missile. In both cases, desired performance was obtained and characteristics of seat performance adequately determined. However, parachute recovery was unsuccessful. Nevertheless, the program

is continuing and it is felt that sufficient knowledge has been gained in these tests to insure future successful supersonic recoveries.

With a reliable recovery system, this type of missile should provide an inexpensive and reliable method of testing aircraft components under operational conditions at supersonic speeds.

## Generation of High Gas Pressure Through Hydraulics

(Continued from page 448)

of this equipment is its flexibility and versatility. The performance of a given unit can be expanded or contracted with relative ease in accordance with changing requirements. A given design can be conveniently expanded to yield higher quantities of high pressure gas by adding an additional gas receiver and additional transfer barriers. Performance can be further improved by increasing the GPM output of the hydraulic system. As a contrast, a pressure booster unit can also be run on "half its cylinders" by valving off a section of transfer barriers and a receiver. This can be done for high pressure gas charging problems not requiring the capacity of the pressure booster unit.

The high pressure gas booster system described in this paper has been in use at the Patrick Air Force Base with success. Maintenance has been low and efficiency high. Many new systems currently being designed indicate the desirability of generating high pressure gas at large flow capacities through the use of hydraulics.

## Gas Temperatures in Liquid Propellant Flames

(Continued from page 453)

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# Contribution of Chemical Reactions to a Time Lag in Nitromethane Rocket Motors

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The effect of oxygen in the thermal decomposition of nitromethane at approximately 250 psia initial pressure and 355 C was studied. Oxygen does not influence the rate of disappearance of nitromethane, but reduces considerably the concentration of hydrogen cyanide and other intermediates. On the basis of a correlation of these results with the fact that oxygen promotes sustained combustion in nitromethane rocket motors, the existence of a time lag caused by chemical processes is proposed.

## Introduction

SEVERAL investigators have dealt recently with the problem of combustion instability in rocket motors. Their theories differ in many important aspects, but most of them relate combustion instability to the time delay that occurs between the injection of the propellants and the liberation of all the thermodynamically available heat (1, 2, 3).<sup>2</sup> It is the purpose of this paper to present experimental evidence that, in addition to the delay attributable to physical processes such as atomization, evaporation, and heating, the chemical transformations may also be responsible for an appreciable time lag. The study to be presented deals with the thermal decomposition of nitromethane at initial pressures of 200 to 300 psia and a temperature of 355 C, and with a comparison of these results with data on the known behavior of nitromethane in rocket motors.

At this point some remarks should be made concerning the validity of correlating data obtained in a laboratory investigation with a study involving an actual rocket motor, especially with respect to differences in pressure and temperature. The initial pressures of nitromethane used in the present study are very close to those employed in rocket motors, but in regard to the temperature there is a difference of about 2000 C between the laboratory experiments and the rocket chamber. It is believed, however, that a significant portion of the chemical reactions may actually occur while the gas is being heated up to the chamber temperature. For instance, the time required to decompose nitromethane to the extent of 30 per cent is only a 0.13 millisecond at 750 C and 0.04 millisecond at 800 C.<sup>3</sup> It may be fair to assume, therefore, that the experimental re-

sults can be correlated with temperatures that are only 400 to 500 C higher, and that one may not have to extrapolate to the equilibrium chamber temperature.

The use of nitromethane as a monopropellant in a rocket motor is limited to some extent by the high  $L^*$  (characteristic length) or high chamber pressure that is required to achieve sustained combustion. However, the addition of only a few per cent of oxygen permits the use of a considerably lower  $L^*$  or chamber pressure. Accordingly, it was decided to investigate the action of added oxygen in the combustion of nitromethane.

## Experimental

A brief outline of the experimental procedure used in the present investigation, which was conducted at 200 to 300 psia initial pressure and 355 C, is as follows: Pyrex ampoules of approximately 3-ml capacity were filled with a weighed amount of nitromethane,<sup>4</sup> frozen in liquid nitrogen, and sealed under vacuum. In some cases a measured quantity of oxygen was also added. Because the volume of the ampoule was known and all experiments were carried out above the critical temperature of nitromethane, its initial pressure could be determined, assuming the validity of the ideal gas laws. The ampoules were immersed in a Blue-Temp<sup>5</sup> heating bath which was held at  $355 \pm 0.5$  C. At the conclusion of a test the ampoule was removed from the furnace and quenched. The quantity of nitromethane that had remained undecomposed was determined polarographically, and the products of the decomposition were analyzed in a mass spectrometer.

## Results and Discussion

Table 1 shows the effect of added oxygen on the rate of decomposition of nitromethane. Oxygen does not affect the rate of disappearance of nitromethane.

Table 1 Effect of oxygen on the rate of decomposition of nitromethane

Initial pressure of nitromethane, 180 to 250 psia; temperature, 355 C

Oxygen mole, %	Duration, min	Amount decomposed, %
..	5	5.1 <sup>a</sup>
5	5	4.3 <sup>b</sup>
15	5	4.9 <sup>a</sup>
..	15	15.2 <sup>b</sup>
13	15	16.5 <sup>c</sup>
..	31	26.9 <sup>d</sup>
6	31	25.6
14	31	27.0

<sup>a</sup> Mean of 5 tests; <sup>b</sup> mean of 2 tests; <sup>c</sup> mean of 3 tests; <sup>d</sup> taken from a plot of initial pressure vs. per cent decomposition.

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<sup>2</sup> Numbers in parentheses indicate References at end of paper.

<sup>3</sup> These figures are based on computations employing the following experimental data: nitromethane at 300 psia initial pressure and a temperature of 355 C decomposes at the rate of 30 per cent in 31 minutes. Extrapolation to the higher temperatures was made by means of the constant,  $k = 10^{11.6} \exp(-53,600/RT)$  sec<sup>-1</sup>, which was determined at subatmospheric pressures, and in the temperature range from 380 to 430 C (4). Satisfactory agreement was obtained between the per cent decomposed (based on an extrapolation to 405 C by this method) and the experimentally determined percentage.

<sup>4</sup> Nitromethane was obtained from the Commercial Solvents Corp. and purified by steam distillation followed by fractional distillation. Its boiling point was 100 C at 750 mm pressure and it contained less than 0.03% water.

<sup>5</sup> Mixture of sodium nitrate and sodium nitrite.

Table 2 Thermal decomposition of nitromethane: product analyses

Temperature, 355 C

Pressure, psia	179	235 <sup>a</sup>	221	310	234
Time	2 min	5 min	15 min	31 min	24 hr
Product	Moles product per mole nitromethane decomposed $\times 100$				
CO <sub>2</sub>	8	19	20	31	34
CO	17	20	18	12	19
CH <sub>4</sub>	9	11	8	5	6
HCN	35	38	39	43	22
CH <sub>3</sub> CN	5	1	3	4	9
C <sub>2</sub> H <sub>5</sub> N	4	2	Trace	..	..
CH <sub>2</sub> O	6	3	1	..	..
NO	42	53	35	26	2
N <sub>2</sub> O	2	2	7	3	..
N <sub>2</sub>	4	2	6	10	35
H <sub>2</sub> O	101	96	100	106	111

<sup>a</sup> Average of three tests.

Table 2 shows the product distribution in the thermal decomposition of nitromethane. Hydrogen cyanide is the major carbon-containing product, and its disappearance is slow compared with that of nitromethane, since after 24 hours, when virtually all nitromethane has disappeared, 22 per cent of the carbon is still in the form of hydrogen cyanide. Nitric oxide, which is the major nitrogen-containing compound, seems to disappear much faster. Table 3 shows the distribution of carbon-containing products that are formed when nitromethane decomposes in the presence of added oxygen. The reason for listing the carbon-containing compounds only and for combining the carbon oxides, is that nitrogen dioxide, which is one of the products in the case of added oxygen, interacts with the mass spectrometer source. This introduces uncertainties in the analyses of nitrogen and its oxides, as well as in the ratio of carbon dioxide to carbon monoxide.

The most significant result of these analyses is that oxygen causes the concentration of hydrogen cyanide, which is the major carbon-containing compound in the thermal decomposition of nitromethane, to be reduced to one tenth its value in the absence of oxygen. At the same time the concentrations of carbon dioxide and carbon monoxide are increased.

The observations that oxygen does not influence the rate of the disappearance of nitromethane and that it affects significantly the product distribution may now be correlated with the fact that the addition oxygen in nitromethane rocket motors promotes sustained combustion. The combustion process is made up of several physical and chemical transformations, and the time periods required for the completion of each of these steps may be symbolized as shown in Table 4. This representation is entirely qualitative, and it is not implied that the time lags are equal in length or that they do not overlap. Because the processes occurring in time lags  $\tau_1$ ,  $\tau_2$ ,  $\tau_3$ , and  $\tau_4$  are of a physical nature, the sum of these time periods are called the physical time lag. During  $\tau_5$  and  $\tau_6$ , however, chemical transformations take place, and the sum of

Table 3 Effect of oxygen on the distribution of the major carbon contained products in the thermal decomposition of nitromethane

Initial pressure of nitromethane: 214 to 250 psia  
Temperature: 355° C  
Duration: 5 min

Product	Nitromethane <sup>a</sup>	Nitromethane plus 16 mole % oxygen <sup>b</sup>
HCN	38	3.5
CH <sub>4</sub>	11	Trace
CO + CO <sub>2</sub>	39	96

<sup>a</sup> Average of three tests.<sup>b</sup> Average of four tests.

these two time lags are identified as the chemical time lag. It is unlikely that the addition of a small amount of oxygen to a nitromethane rocket motor affects any of the steps comprising the physical time lag, and it must be concluded that oxygen influences only the chemical time delay. On the basis of the experiments reported, the action of oxygen appears to be related to that portion of the chemical time lag ( $\tau_6$ ) which represents the transformation of the intermediates into final products. The most important intermediates in the decomposition of nitromethane are hydrogen cyanide and nitric oxide (Table 2). Hydrogen cyanide is endothermic by 31 kcal and is present in large concentrations; its rate of disappearance is also relatively low. Nitric oxide is endothermic by 21 kcal, and is also present in large concentrations, but it appears to be transformed into final products more rapidly than hydrogen cyanide, as shown in Table 2. Whatever the relative effects of these two compounds on time lag  $\tau_6$  may be, it appears certain that hydrogen cyanide is an important intermediate, and that an additive that accelerates its transforma-

Table 4 Distribution of time lags (schematic)

$\tau_1$	$\tau_2$	$\tau_3$	$\tau_4$	$\tau_5$	$\tau_6$
Atomization	Heating of liquid droplets	Evaporation	Heating to temperature at which reaction is appreciable	Reaction to intermediates	Reaction to final products
Physical time lag				Chemical time lag	

tion into final products should aid materially in reducing this time delay.

Additional work must be done to determine the relative contributions of the physical and chemical time lags to the over-all time lag, and it is very likely that in different systems one or the other may predominate. There is also the possibility that even if the chemical time delay turns out to be short compared to the physical delay, its contribution may be just sufficient for the sum of the two lags to exceed a critical value, and thus produce instability.

Summarizing on the basis of the evidence, it is proposed that the presence of oxygen in nitromethane rocket motors serves to remove endothermic intermediates and thus to reduce the chemical time lag.

### Acknowledgment

The author wishes to express his gratitude to the Office of Naval Research for support of the work which led to this paper.

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## General Enthalpy-Temperature-Entropy Diagram

(Continued from page 462)

intermediate enthalpy and entropy values above 2500 K for OH and NO where the original references contain only data for 250° temperature intervals. In this case, a further improvement in the interpolation was made by the addition of a corrective term which was calculated assuming that the error resulting from the supposed linear progress of  $C_p$  was also linear within the interval of interpolation.

For  $\text{CO}_2$ , the enthalpy and entropy values were taken from an advanced copy of (7) and proved to be slightly different from the data of the final (7). The maximum error, however, does not exceed one unit in the fifth significant figure.

### Summary

A diagram has been developed which, extending from room temperature to 5000 K (9000 R), permits the reading of the relative molar enthalpy,  $\int_{T_0}^T C_p dT$ , and the molar temperature function of entropy,  $\int_{T_0}^T \frac{C_p}{T} dT$ , for all bi- and triatomic gases and their mixtures.

The large temperature range and the wide variety of gases for which this general diagram is intended made it impractical to develop an enthalpy-entropy (Mollier) diagram. Separate diagrams for enthalpy versus temperature and for the entropy temperature function were constructed. However, these two diagrams are combined into a single system by plotting them against the same temperature abscissa, and simple graphical corrections are included to permit the reading of internal energy and entropy at constant volume directly from the chart. Applying the simple general relationship for the molar entropy change with pressure,  $\Delta S' = R \ln P_1/P_2$ , or with volume,  $\Delta S'_v = R \ln V_2/V_1$ , these two systems can be

used simultaneously to fulfill the functions of a true Mollier diagram for most practical purposes.

Values for individual gases are obtained by attributing to each gas two individual parameters. The diagram was constructed in such a way that for the three most common gases,  $\text{N}_2$ ,  $\text{H}_2\text{O}$ , and  $\text{CO}_2$ , the parameters are constants and have simple numerical values. For other bi- and triatomic gases, as well as for many higher atomic gases, the parameters are nearly constants over the major part of the temperature range. For gas mixtures, the parameters are quickly determined as the weighed averages of the component parameters. For gas mixtures composed chiefly of bi- and triatomic gases, constant parameters can be assumed in almost all practical cases.

Use of the diagram is limited to ideal gases defined by  $(\partial H/\partial P)_T = 0$  and to mixtures of ideal gases of invariant composition. The latter limitation requires, if dissociation has to be considered, that the composition of the gas be calculated by separate methods and that "frozen composition" be assumed within a cycle calculation. Apart from these two limitations, the values obtained from this diagram are not approximations but true values for enthalpy and entropy including effect of temperature variable specific heats of individual gases.

The correct parameter functions are presented for the gases  $\text{H}_2$ ,  $\text{CO}$ ,  $\text{O}_2$ ,  $\text{OH}$ ,  $\text{NO}$ ,  $\text{H}$ , and  $\text{O}$ . For other gases, they can easily be calculated from their actual enthalpy and entropy values.

The method is derived and use of the tables and diagrams is explained. The determination of a rocket nozzle configuration and the quick calculation of the specific impulse for high chamber pressure are discussed as examples.

### Acknowledgments

The cooperation of the U. S. Naval Ordnance Test Station enabling the distribution of full-sized copies of the diagrams is highly appreciated.

The author also wishes to express his gratitude to Dr. John Sinnott for valuable suggestions in the preparation of the manuscript, and to Dr. Joseph Hilsenrath, National Bureau of Standards, Thermodynamics Section, who made available advance copies of the thermodynamic functions of  $\text{CO}_2$  and  $\text{H}_2\text{O}$ , and to Mr. George Hirvela of Trend Engineering and Publications, Inc., Los Angeles, Calif., for the excellent work done on drawing the diagrams.

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# Technical Notes

## Thermodynamic Properties of Gas at High Temperatures and Pressures

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### 1 Equation of States of Dense Gas

WHEN the density of gas is high, it is well known that the simple equation of states for a perfect gas can no longer be expected to be valid. The most crude approximation to the equation of states for a dense gas is that of Van der Waal. If  $P$  is the pressure,  $v$  the volume per molecule,  $T$  the temperature, and  $k$  the Boltzmann constant, then the Van der Waal equation is

$$\left(P + \frac{a}{v^2}\right)(v - b) = kT \quad [1]$$

where  $a$  and  $b$  are two constants, small in magnitude. The constant  $b$  is usually simply identified as four times the volume of a molecule. If the molecules are assumed to be spheres of diameter  $D$ , then

$$b = \frac{2\pi}{3} D^3 \quad [2]$$

At high temperatures, the density of gas can be large only if the pressure is very high. Then the term  $a/v^2$  is not important in comparison with  $P$ , and Equation [1] can be simplified into the so-called covolume equation of states

$$P(v - b) = kT$$

Or we can write

$$\frac{Pv}{kT} = 1 + \frac{1}{\frac{3}{2\pi} \frac{v}{v^*} - 1} \quad [3]$$

where  $v^*$  is a volume defined by

$$v^* = D^3 \quad [4]$$

However, because of the crude approximation in the Van der Waal equation of states, neither Equation [1] nor Equation [2] can be expected to be sufficiently accurate for gas at very high temperatures and high pressures. An example of such state of matter is the gaseous products of detonation of condensed explosives, where temperatures of several thousand degrees Kelvin and densities of the order of solids occur. Other more elaborate equations of states designed to cover the whole range of pressures and temperatures suffer from the same defect.

For products of detonation of condensed explosives, a more accurate equation is the Halford-Kistiakowsky-Wilson equation (1)<sup>2</sup>

$$\frac{Pv}{kT} = 1 + KT^{-1/4} \exp(0.3 KT^{1/4}) \quad [5]$$

where

$$K = \sum_i n_i K_i \quad [6]$$

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<sup>2</sup> Numbers in parentheses indicate References at end of paper.

$n_i$  is the number of moles of the  $i$ th molecular species per unit volume, and  $K_i$  is the empirical constant for the  $i$ th species. For water and ammonia, the  $K_i$ 's are 108 and 164 cm<sup>3</sup> per mole, respectively. It is suggested that when  $K_i$  is not known, it can be calculated as

$$K_i = 5.5 \left(\frac{2\pi}{3}\right) ND_i^3 \quad [7]$$

where  $N$  is Avogadro's number, and  $D_i$  is the low energy collision diameter of molecules of the  $i$ th species.  $K_i$  is thus equal to 22 times the volume of one mole of spherical molecules of diameter  $D_i$ . If there is only one species of molecules, then Equations [6] and [7] give

$$K = 5.5 \left(\frac{2\pi}{3}\right) \left(\frac{t}{v^*}\right) \quad [8]$$

where  $v^*$  is a volume defined by Equation [4]. Thus for a single gas, the Halford-Kistiakowsky-Wilson equation can be written as

$$\frac{Pv}{kT} = 1 + \frac{11.51}{\left(\frac{v}{v^*}\right) T^{1/4}} \exp\left(\frac{3.453}{\left(\frac{v}{v^*}\right) T^{1/4}}\right) \quad [9]$$

It is easily seen from Equations [5] and [9] that they are an improvement over the covolume equation [3]; now the "compressibility,"  $Pv/kT$ , is unity as  $T \rightarrow \infty$  even with  $v/v^*$  finite. This result is to be expected on the general ground that molecules are never rigid spheres, but "squeezable," i.e., closer approach is possible if two molecules collide with greater kinetic energy. At very high temperatures, the kinetic energy of the molecules is very high, then the effective size of the molecules in collision must be very small. Now the effects of gas imperfection is proportional to the molecular size. Therefore as  $T \rightarrow \infty$ , the effective molecular size vanishes and the gas becomes a perfect gas, even if the volume ratio is finite. Hence really satisfactory equation of states must have this property. This is so for the Halford-Kistiakowsky-Wilson equation.

But even when the constant  $K$  is related to the low energy collision diameter of the molecules through Equations [7] and [8], still no proper account is made for the strength of molecular interaction. If we represent the interaction between a pair of molecules by the Lennard-Jones potential  $\epsilon(r)$ , a function of the intermolecular distance  $r$

$$\epsilon(r) = 4\epsilon^* \left[ \left(\frac{D}{r}\right)^{12} - \left(\frac{D}{r}\right)^6 \right] \quad [10]$$

then besides the collision diameter  $D$ , there is the parameter of equilibrium potential  $-\epsilon^*$ , the potential when the pair of molecules are at their equilibrium distance  $2^{1/6}D$ . A convenient parameter for  $\epsilon^*$  is the "characteristic temperature of interaction,"  $\Theta_i$ , defined as

$$\Theta_i = \frac{\epsilon^*}{k} \quad [11]$$

Then we expect the equation of states to be of the form

$$\frac{Pr}{kT} = 1 + f(T/\Theta_i, v/v^*) \quad [12]$$

i.e., the compressibility should be a function of temperature ratio  $T/\Theta_i$ , but not temperature itself as is in the Halford-Kistiakowsky-Wilson equation. Furthermore, according to our argument on vanishing imperfection at  $T \rightarrow \infty$

$$f(\infty, v/v^*) = 0 \quad [13]$$

With the general concept that the equation of states for dense gas must satisfy the conditions embodied in Equations [12] and [13], many of the equations of states proposed by various authors can be ruled out as unreliable. For instance, Cottrell and Paterson suggested (2) that

$$\frac{Pv}{kT} = 3 + \frac{\text{const}}{(T/\Theta_1)^{1/2} \left( \frac{v}{v^*} \right)} \dots \dots \dots [14]$$

Since the condition of Equation [13] is not satisfied, this equation cannot be reliable for very high temperatures. In a similar manner, the theoretical equation of states proposed by Zwansig (3) is also unacceptable, because he makes the untenable assumption that at very high temperatures the molecules interact as rigid spheres.

## 2 Lennard-Jones and Devonshire Theory

If there were sufficient experimental data for gas at very high temperatures and pressures, we can try to fit the data to the nondimensional equation of states of Equation [12] and determine the function  $f(T/\Theta_1, v/v^*)$ . This is, in effect, the application of the principle of corresponding states. Unfortunately, except for two old measurements on hydrogen and helium by Bridgman (4), no experimental data are available at high enough temperatures and pressures for this to be possible. We must then turn to theory to determine the proper equation of states. If we are interested in the region of density where the nondimensional volume  $v/v^*$  is near unity, i.e., densities of the order of liquid density at low pressure, then the Lennard-Jones and Devonshire theory of liquids and dense gas is a very good approximation to the true physical situation. Here the main defect of not allowing for empty lattice sites in the theory is not important. Furthermore, Wentorf, Buhler, Hirschfelder, and Curtiss (5) have carried out very extensive and accurate calculations for the Lennard-Jones and Devonshire theory. We can use their tabulated values directly. But for practical calculations, it would be more convenient to put their result in a simple analytic form for easy interpolation and extrapolation. This is the main purpose of the present note.

Of course theoretically the equation of states can be directly obtained from the formulation of Lennard-Jones and Devonshire theory without having to use the numerical results. But the interested range of parameters  $T/\Theta_1$  and  $v/v^*$  is such that no simple analytic result can be obtained. This is quite different from the case of liquid state where  $T/\Theta_1$  is small enough for the simple analytic treatment (6) to be successful.

By using the tabulated values of  $Pv/kT$  given by Wentorf and collaborators, we can plot  $T/\Theta_1$  against  $v/v^*$  for constant  $Pv/kT$ . The result of this preliminary investigation indicates that instead of two variables  $T/\Theta_1$  and  $v/v^*$  in  $f$  of Equation [12], a single variable  $\eta$  is sufficient with

$$\eta = (T/\Theta_1)^{1/6} (v/v^*) \dots \dots \dots [15]$$

Then, according to Equations [12] and [13]

$$\frac{Pv}{kT} = 1 + f(\eta), \quad f(\infty) = 0 \dots \dots \dots [16]$$

In Fig. 1,  $1/f = (Pv/kT - 1)^{-1}$  is plotted against  $\eta$ . We see that the points are grouped around a straight line for  $T/\Theta_1 = 10, 20, 50, 100$ , and 400 and  $v/v^*$  near unity. In fact, we find the approximate equation of states for gas at very high temperatures and pressures as

$$\frac{Pv}{kT} = 1 + \frac{1}{0.278\eta - 0.177} \dots \dots \dots [17]$$

The condition of Equation [13] is thus satisfied. This result is accurate to about 10 per cent.

Although the equation of states, Equation [17], is obtained for dense gas at high temperature, it may be interesting to see how well it behaves for dilute gas. For dilute gas,  $v/v^*$  is

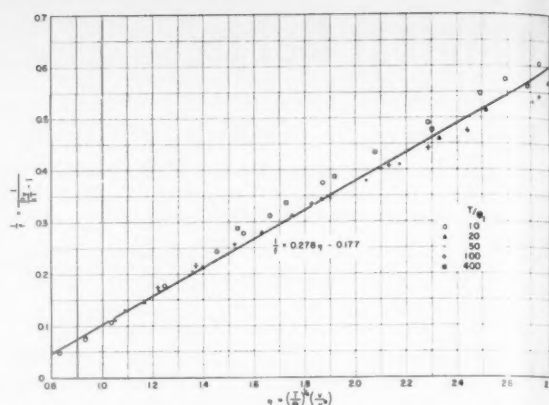


Fig. 1

very large, then  $\eta$  is very large. Therefore for dilute gas at high temperatures, we have

$$\frac{Pv}{kT} \cong 1 + \frac{1}{0.278\eta} = 1 + 3.594(T/\Theta_1)^{-1/6} (v/v^*)^{-1} \dots [18]$$

On the other hand, the exact equation of states for dilute gas at very high temperatures is given by the virial equation, retaining only the leading term in the expansion of the second virial coefficient (7). Thus

$$\frac{Pv}{kT} \cong 1 + \frac{2\sqrt{2}\pi}{3} \Gamma\left(\frac{3}{4}\right) (T/\Theta_1)^{-1/4} (v/v^*)^{-1} = 1 + 3.630(T/\Theta_1)^{-1/4} (v/v^*)^{-1} \dots [19]$$

By comparing Equations [18] and [19], it is seen that aside from the difference in the exponent of  $T/\Theta_1$ , our approximate equation of dense gas at high temperatures even reproduces some of the characteristics of a dilute gas. Therefore for rough approximations, our equation of states, Equation [17], can be even used for the complete range of  $v/v^*$ . Of course, Equation [17] can be true only for high temperatures, say  $T/\Theta_1 > 10$ .

## 3 Other Thermodynamic Functions

Since the proposed equation of states is approximately true for even dilute gas, we may use it to calculate other thermodynamic functions by integrating to  $v/v^* \rightarrow \infty$ . For instance, if  $E$  is the energy per molecule, then the general thermodynamic law states that

$$\left( \frac{\partial E}{\partial v} \right)_T = T \left( \frac{\partial P}{\partial T} \right)_v - P$$

Thus, by differentiating Equation [17], using Equation [15]

$$\left( \frac{\partial E}{\partial v} \right)_T = - \frac{kT}{v} \frac{0.278\eta}{(0.278\eta - 0.177)^2} \dots \dots \dots [20]$$

Thus, if  $E_\infty$  is the energy per molecule when  $v/v^* \rightarrow \infty$ , i.e.,  $E_\infty$  is the energy per molecule calculated for a perfect gas, then the energy per molecule due to imperfection of the gas is

$$(E - E_\infty)_T = - \int_v^\infty \left( \frac{\partial E}{\partial v} \right)_T dv \dots \dots \dots [21]$$

By substituting Equation [20] into Equation [21], and by considering  $(E - E_\infty)/kT$  to be a function of  $\eta$  only, we obtain

$$(E - E_\infty)_T = \frac{kT}{6} \frac{1}{0.278\eta - 0.177} \dots \dots \dots [22]$$

We observe that although the result involves the use of our approximate equation of states for large  $v/v^*$ , the contribu-

(Continued on page 478)

# Jet Propulsion News

Alfred J. Zaehring, American Rocket Company, Associate Editor

## Rockets and Guided Missiles

**N**IKE, during its cold weather tests at Fort Churchill, Canada, proved to be very effective. Tests included roadability, preparation of launching sites, moving equipment into action, and electronic checkout. Tests were begun in November 1954 and were completed in March 1955. The operation was carried out by 49 trained Canadian personnel, with U. S. and Canadian observers present. A typical NIKÉ site here in the United States occupies an area of about 50 acres and costs about \$1.0 million. The 20-ft long missiles and boosters weigh over a ton each and are stored in concrete vaults 24 ft underground. Each NIKÉ site is said to be about as safe as a gas station due to stringent safety practices. Once a year, the crews go to White Sands where practice tests are made. Each NIKÉ battery is composed of six officers, two warrant officers, and 101 enlisted men. An additional \$160 million is being appropriated in 1956 for NIKÉ installations. However, it was stated by the Senate Armed Services Committee that within a few years NIKÉ may be outmoded by newer long range systems now under development and that the NIKÉ system can be adjusted to take care of new developments.

- The Army has officially stated that progress is being made on an improved surface-to-air missile which is designed for low-altitude air defense. It was also stated that an improved tactical support missile is in the works and work is continuing on a longer range missile which can use new and improved warheads.

- Two guided missiles are being developed by Cornell Aeronautical Laboratory, Buffalo, N. Y. One is a surface to-surface ground-support missile for the Army and Marine Corps, called LACROSSE. LACROSSE is to be produced by Glenn L. Martin and 20-100 missiles are to be produced initially. The other Cornell-developed missile is for air-to-air purposes. Prototypes have been tested and production



W. L. Mazson Corp.

New missile launchers for Navy show heavy-duty construction



Aerofet-General

Instrumentation crew readies new Aerobee-Hi. First shot hit 125 miles, others 135 miles, latest 180 miles

is being sought. Another Cornell missile project is concerned with the problem of combating enemy missiles and missile vulnerability in general.

- FALCON rocket, GAR-98 is now undergoing modification. Among others, its radome-ogive is being streamlined. Development of the solid propellant system for FALCON was by Thiokol Chemical Corp. in coordination with U. S. Air Force, U. S. Army Ordnance Corps, and Hughes Aircraft Co. Meanwhile, it has been hinted that our air weapons might soon be armed with nuclear warheads. It has been found feasible to fit an atomic warhead in a projectile as small as 4 or 5 inches in diameter. Other atomic weapons have also been reduced in size and bulk. With the prospect that a megaton hydrogen weapon can be carried by a single fighter-bomber, it appears that our longer-range rockets may have no difficulty in carrying these warheads.

- Boeing IM-99 missile has been successfully fired at the long-range proving ground of the U.S.A.F. at Cocoa, Fla. The IM-99 is known as BOMARC and takes its name from Boeing, BO, and the Michigan Aeronautical Research Center, MARC. The latter group, under the aegis of the University of Michigan, is now known as the Willow Run Research Center and began studies on defense weapons systems and later built research models of ground control equipment for Boeing. The BOMARC project involved not only design and development of the missile itself but also allied control, ground handling, and logistical equipment for the complete defensive system.

- WIZARD is the name of an anti-aircraft missile by the Willow Run Research Center of the University of Michigan. Studies on this missile led to the Boeing GAPA program and later evolved into the BOMARC missile.

- The BULLPUP missile by Martin has grown up and will be called BULLDOG. The Navy BULLDOG, an air-to-surface missile, is propelled by a solid propellant rocket and is 11 ft long with a diameter of 1 ft.

- DAN is a new two-stage rocket which can hit the 70-mile altitude level. ARDC has made ground firings at Wallops Island, Va. DAN uses the Nike booster as the lower stage and the 3.5DS-7500 Deacon rocket as final stage. Aim of the ARDC program is 100 miles for upper atmosphere soundings. The 70-mile altitude is probably a record for an all-solid propellant rocket system.

**EDITOR'S NOTE:** The information reported in this Section has been selected from approved news releases originating with the Department of Defense, private manufacturers, universities, etc., and from published news accounts in journals and newspapers. The reports are considered generally reliable, although no attempt has been made to verify them in detail.

● A new liquid propellant rocket motor was disclosed by the de Havilland Engine Co., of England. Called the "Spectre," the motor is designed for the propulsion of manned interceptors. The Spectre has undergone extensive static testing, and flight tests are expected soon.

● The SFECMAS SS10 is a new French wire-controlled guided rocket for air-to-air or ground-to-air use.

## Made in U.S.A.—A Lineup of Liquid Propellant Rocket Engines



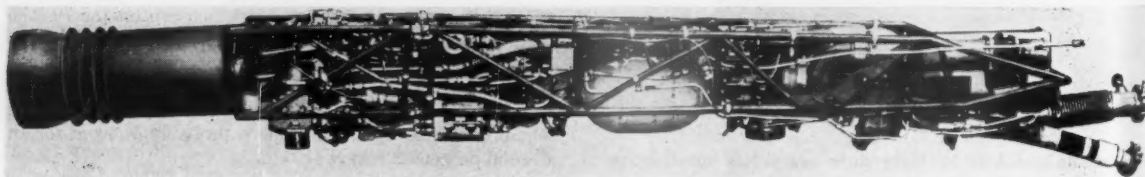
RMI

Viking rocket engine (right) is compared with V-2 (left). V-2 developed 56,000 lb thrust, Viking 20,000 lb. However, the RMI-developed engine develops nearly twice as much thrust per pound of engine weight and is only about one third the size of the V-2 engine.



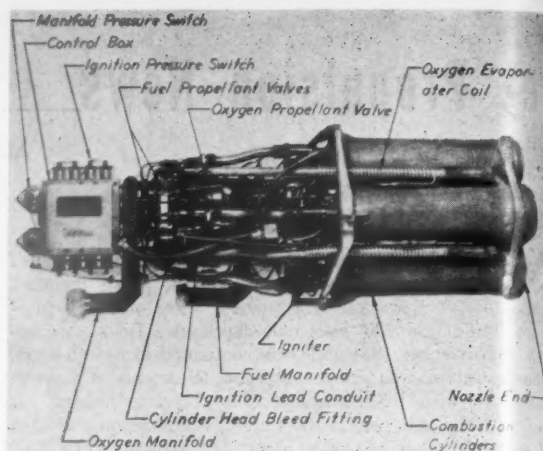
Ryan

Production line of Corporal rocket engines. Welded construction is featured.



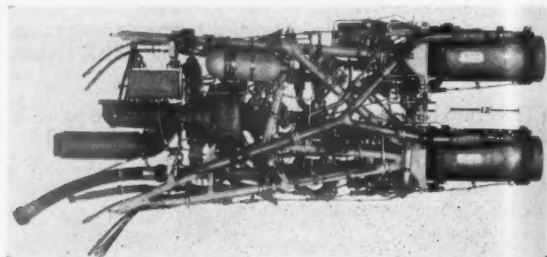
Aerogel-General

Model AJ24-1 (YLR45-AJ-1). A two-thrust chamber, bipropellant rocket engine with related equipment and controls mounted on a retractable frame. Designed as a permanently installed ATO unit for the B-47B jet bomber.



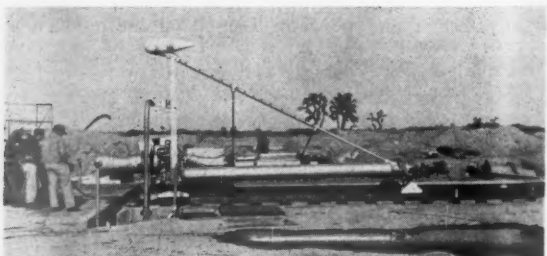
RMI

Model 1500N4C. 4 cylinder, regeneratively cooled. Pressure feed. 75% ethyl alcohol fuel, liquid oxygen oxidizer. Specs: max. diam, 19 in; length 56 in.; frontal area, 2 sq. ft; weight, 210 lb; Propellant consumption, 0.0052 lb/lb thrust/hr; specific impulse, 192 sec; thrust, 6000 lb. Powered Bell X-1, Douglas Skyrocket D-558-2, and Republic XF-91.



Aerogel-General

LR63AJ-1. Single thrust chamber with related equipment and controls mounted on a welded tubular frame. Designed as a permanently installed ATO unit for F-84 jet fighter.



North American

Sled Motor. Can deliver 50,000 lb thrust for 8 sec. Liquid oxygen-alcohol. Motor is seen at rear, long cylinder is for propellant. Used at Edwards AFB, California, to accelerate 4,000 lb sled from dead stop to over 1500 mph in 4.5 sec while traveling 5500 ft. Fuel cost per run: \$100.



# Space Flight Notes

Kurt R. Stehling, Bell Aircraft Corporation, Contributor

## Historical Development

THE President's announcement on the establishment of a satellite for scientific purposes has sharpened interest in this subject, particularly in those proposals which have appeared in various journals during the past few years.

No proposed enterprise in the history of our contemporary technology has been so well documented before the event has occurred as has that of "space flight." The speculation in this field, which began long before rocket propulsion was known, was greatly increased in scope and magnitude with the advent of the great rocket and "astronautical" pioneers, Goddard, Oberth, Esnault-Pelterie, Sänger, Rynin, von Pirquet, and others. Most speculation before World War II discussed flight into outer space, i.e., to the Moon, planets, and beyond. Oddly enough, the development of rocket propelled vehicles during World War II changed the objectives of many astronauts from "interplanetary" travel to the "satellite" flight. Perhaps this was not so strange; the development of the V-2, among other vehicles, showed the potentialities of such vehicles while emphasizing their limitations and need for much further development.

When the engineering (and financial) limitations of rocket propelled missiles were comprehended the various articulate workers in the field, using as a sounding board scientific meetings and journals, presented technical papers of often considerable scientific merit. One of the best examples is the paper by Seifert, Malina, and Summerfield in the *American Journal of Physics*, vol. 15, nos. 1, 2, 3; 1948. The relatively modest objective of satellite flight was treated, in great detail, as a prelude to the grander objective of interplanetary travel. The subject of interplanetary travel was, of course, not forgotten; indeed many articles appeared on this subject. However, only a few of these were detailed and rigorous studies (e.g., von Braun's "The Mars Project") or discussions of new methods of propulsion suitable for flight in space—such as electron, nuclear particle, or photon propulsion. These serious studies of lunar or planetary travel all recognized the need for orbital or satellite vehicles as refueling or stopover stations for the space vehicles. Other workers thought only of the satellite as an immediate goal with the objective of extraatmospheric research which would extend the existing knowledge of cosmic ray particles, the nature and extent of the earth's magnetic field, the extent of the sun's radiation spectrum, and the concentration (or density) of the atmosphere at altitudes beyond 200 miles. This preoccupation with the scientific uses of a satellite resulted first from the necessity of avoiding any military connotation because of security restrictions, which affect most of the writers on this subject; secondly, it has been the hope of many of the scientists and engineers who are working in the guided missile industry that some nonmilitary applications (however necessary these are at present) would be found for their products. The scientific satellite seems to be such an application.

Various papers on the subject of satellite and space travel will then be reviewed briefly, breaking the subject down into three major segments—with the emphasis on the small satellite:

- 1 Space travel; i.e., to the Moon and planets.
- 2 The description of large space "stations" and orbital refueling bases.
- 3 Small earth satellites with descriptions of their construction and utility.

EDITOR'S NOTE: This new section of the JOURNAL is inaugurated in this issue as a regular monthly feature in response to many requests from ARS members. Mr. Stehling will offer news and comments on space flight and on such related topics as upper atmosphere research, astronomical findings, and aeromedical developments. From time to time, articles and lectures that have appeared elsewhere will be reviewed. Suggestions for subjects worthy of presentation will be welcome.

## Space Travel

*The Mars Project*, by W. von Braun, The University of Illinois Press, 1953.

In essence, the "project" consists initially of approximately 13,000 ton weight three-stage returnable ferry vessels which would lift the vehicles and 70 men for the Mars expedition to an orbital take-off path 1075 miles above the earth. A flotilla of 10 space ships weighing some 4000 tons will then approach Mars, via an elliptical orbit. The flotilla will enter a captive orbit around Mars and will release three rocket-aircraft type landing "boats," which will deposit 50 men on the planet's surface. These hardy pioneers will then live for 400 Martian days in a presumably noisome environment. Then, they will rejoin their lonely 20 companions who have been orbiting about Mars all this time. Leaving their landing boats and other miscellaneous hardware behind them, the 70 pioneers will return to a circum-earth orbit from which they will finally glide to Earth after an absence of almost three years.

The author makes a number of general statements which do not have a substantial premise. He says that he cannot foresee a practical application of nuclear rocket propulsion within the "next 25 years" which would be superior to his nitric acid-hydrazine system. It is true that nuclear rocket propulsion from the earth's surface may not offer an advantage over chemical propulsion; yet surely, in the time that it would take to develop a 13,000-ton project, as described in the book, it would be possible to develop a nuclear rocket capable of interplanetary navigation.

Comment: For the future.

## Large Space Stations

The above writers and others who have considered lunar and interplanetary travel have recognized the need for so-called orbital tankers or refueling stations which would replenish the fuel so prodigally expended by any rocket vehicle leaving the earth. These orbital tankers are necessarily manned vehicles circling in an orbit about the earth until their store of fuel is exhausted. Other vehicles of comparable size were considered as manned space stations or satellites containing a crew of scientists which would maintain a perpetual vigil as an outpost of science. A battery of scientific instruments would permit these men to make many astrophysical measurements which are impossible on earth. The favorite shape is the "doughnut" or cartwheel which would house the personnel in the outer rim; the vehicle is then rotated or spun so that angular acceleration obtained this way would replace the pull of gravity which is "cancelled" by centrifugal force of the vehicle orbiting around the Earth.

Again, a great amount of popular literature has appeared on this subject, without going into much detail. Below is listed a paper which has more detail than is contained in the popular literature.

*A Preliminary Design Study of a Three Stage Satellite Ferry Rocket Vehicle with Piloted Recoverable Stages*, by D. C. Romick, R. E. Knight, and J. M. Van Pelt, Goodyear Aircraft Corporation. ARS Preprint No. 186, 1954.

A preliminary design study is presented of a three-stage ferry rocket vehicle for carrying a sizable payload into a satellite orbit, and returning to the Earth's surface. Each separate vehicle in this system has delta wings and nose doors which close upon separation from the next stage, and an ex-

tendable landing gear. Thus, each vehicle, when its main function is completed, converts to a glide-type aircraft, and its crew guides it to a landing at a base where jet engine pods and other minor parts are installed so that the crew may then fly it back to the launching base as an airplane. The design study claims technical feasibility of the entire vehicle configuration, as well as piloted recoverable stage principle, and also demonstrates operational, logistical, and other advantages.

The authors use the sound premise that the boosters of a satellite vehicle or other space vessel be returnable. They have worked out some of the weights and sizes for giant space ships weighing 9000 tons or more. Professor G. A. Crocco, University of Rome, has also discussed the desirability of winged, returnable boosters although without going into the hardware detail which these authors have done.

**Comment:** The concept of recoverable winged boosters has merit. However, when the difficulties of designing and building even small rocket aircraft are considered, then the planning of 9000 ton space vessels must perforce be left to our distant progeny.

### Small Earth Satellites

These proposals have been discussed only in order to show the great difference in magnitude and scope between them and the "minimum" scientific satellite which is currently very much in the news. The terms "minimum," "baby," or even "small" have been widely used to describe an orbital vehicle with a payload from "nothing at all" to perhaps a hundred pounds or so of instrumentation. The empty satellite is certainly a minimum vehicle and would serve perhaps only as a monument to its builder, although it could perchance yield a little information on orbital stability and residual air drag by the characteristics of its flight path. A good criterion for the least useful payload might be that of the instrument weights of available small sounding rockets such as the "Deacon" or the earlier WAC Corporal. This would mean a payload of perhaps 30 to 50 pounds.

The subject of the small scientific satellite has been an attractive one for some space flight protagonists; indeed, it seems a logical and feasible step toward space flight to those who are close to the missile industry. A beginning has to be made sometime—and a small satellite feasible within the domain of present missile technology will serve as a catalyst which will leave the present necessary military dependence of missiles at a time when this dependence may be reduced by a change in the world situation.

Several unclassified papers have appeared in this country and in Europe by people who wish to see some beginning made on space flight, without diffusing their thinking and speculation on grander objectives. Some of the papers, such as those by G. A. Crocco in Italy, D. F. Lawden in England, and K. A. Ehricke in the United States, have considered small satellites in terms of the possible orbits, utility as carriers for other vehicles, and returnability of the various stages.

S. F. Singer and W. von Braun, and others have discussed in detail the payload and uses of a miniature satellite without specifying possible hypothetical rocket vehicles which could lift the satellite into its orbit.

J. R. Pierce, I. S. Bowen, H. E. Newell, E. Bollay, J. O'Keefe, and H. J. Schaefer have emphasized various scientific enterprises for an orbital vehicle without discussing either the final vehicle or its carriers. Their papers formed an appendix for the proposal of the ARS Space Flight Committee to the National Science Foundation [JET PROPULSION, vol. 25, February 1955, p. 72].

J. H. Wyld, K. R. Stehling and R. M. Missert, and P. E. Sandorff have shown possible new approaches to launching a satellite based either on prognostication of future vehicles or the availability of existing unclassified hardware.

**The Crucial Problem in Astronautics: Recovery of Multistage Vehicles**, by G. A. Crocco, University of Rome, Italy; JET PROPULSION, vol. 24, September-October 1954, p. 313.

Professor Crocco has presented an interesting problem in Astronautics—The Recovery of Multistage Vehicles. From his long experience in practical aeronautical development he argues the advantages of a cautious step-by-step program of developing and proof-testing multistage orbital vehicles. In his view, each stage must be a complete flying machine capable of ascending into space and returning safely to earth.

This view is a sound one if a winged-rocket satellite is considered. The small scientific satellite treated here might be inexpensive enough, however, to permit stages I and II to be freely dropped to earth or into the ocean. The possibility of special recovery devices such as ribbon parachutes might be considered if the development of such devices does not become more expensive than the vehicle itself.

**A New Supply System for Satellite Orbits**, by Krafft A. Ehricke, Convair; JET PROPULSION, vol. 24, September-October 1954, p. 302.

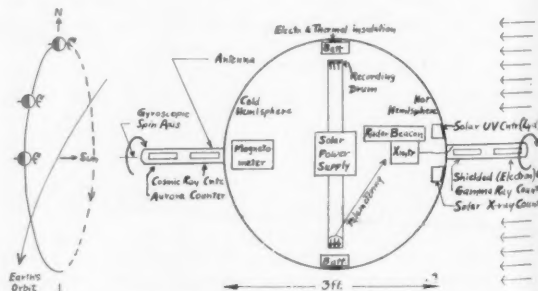
In this and other papers Ehricke details the flight paths, weights, and logistics for various space vehicles. He stresses the engineering aspects of orbital operations and analyzes the existence of optimum satellite orbits for departure and arrival of interplanetary expeditions.

Ehricke has been very thorough in his analysis of the payload requirements for various orbital vehicles and the transfer of such payload to other, already established satellites. He always weaves into his works the ultimate application of space flight and considers satellites largely as means to this end. Therefore his theses, while quite erudite, are of long-range interest and point out no immediate specific method of bringing a small vehicle into an orbit.

**A Minimum Orbital Unmanned Satellite of the Earth (MOUSE)**, by S. F. Singer, University of Maryland. ARS Preprint No. 195-55.

The writings of Dr. Singer have reached much prominence during the last two years. He has attracted this attention because of the perspicacity of his proposals and the neat and suggestive name for his proposed vehicle MOUSE, meaning "Minimum Orbital Unmanned Satellite Earth." This name was proposed in collaboration with Mr. A. C. Clarke and Mr. A. V. Cleaver of the British Interplanetary Society.

Although the idea or concept of the minimum satellite did not originate with Singer, he has managed clearly to specify the type and use of measuring instrumentation which could be incorporated into less than 100 lb of payload carried aloft into an orbit.



The MOUSE - Minimum Orbital Unmanned Satellite, Earth  
Showing Gyroscopic Stabilization Scheme & Solar Power Supply  
and Astrophysical Instrumentation

**MOUSE.** A small spheroid, about as big as a basketball, spinning about a polar axis. This axis consists of two radiation counters protruding from either end. Their cylindrical shape and symmetrical electrode arrangement permit spinning while still producing a readable signal.

He discusses the gathering of such data as the nature and extent of the Sun's radiation, the quality and distribution of cosmic ray primaries, the distribution of the earth's magnetic field, and the albedo or reflective power of the earth.

Power could be supplied by a Bell Telephone Laboratory

silicon solar battery which would relieve the vehicle of its burden of batteries and supply electric power indefinitely. MOUSE is usually pictured as a spheroid rotating or spinning about a polar axis, which consists of photon or other counting tubes projecting beyond the body. The spinning of the vehicle imparts to it a measure of orientation stability such as is achieved with a spinning bullet.

It is easily possible, of course, to be too optimistic on the quantity and reliability of instrumentation which could be carried by a small vehicle. The operational reliability of components is compromised when these exceed some arbitrary limit of interdependence, numbers per unit volume, and nature of the intelligence which the components must convey. Therefore it seems that unless miniaturization and reliability are advanced beyond present limits, MOUSE ought to have at first a more modest objective, perhaps only the counting of cosmic ray particles.

At any rate, MOUSE sums up nicely the capabilities and characteristics of a small orbital vehicle. However, if the proposal is to be translated into reality, the onerous problems of component packaging and the method of spin stabilization must be solved, to say nothing of the problem of placing MOUSE into its (her?) orbit.

*The Baby Space Station*, by W. von Braun, Redstone Arsenal, and Cornelius Ryan, *Colliers*, June 27, 1953.

This lavishly illustrated article, which represents a considerable shrinking of von Braun's earlier horizons, describes a small or medium-sized orbital vehicle circling about the earth for 60 days, at 200-mile altitude before residual air drag brings it back to the earth's atmosphere, to burn up like a meteor. The illustrations of a world-wide system of telemetering receiving stations, of a mighty multistage rocket before take-off and the vehicle itself, first shown in section with several nervous monkeys floating about inside and then later shown burning in the atmosphere, are truly awe-inspiring. One can only hope that our Congressional representatives who must vote the money for satellite studies are not intimidated by the size of this enterprise; furthermore, are the S.P.C.A. people impressed by the humane provisions which have been so thoughtfully included for the monkeys' benefit? Nothing is too good for the primates—they have self-flushing sanitary facilities, fresh air, and a delicious cuisine—everything is done to surround them with the environment of dignity and gentility to which they have been accustomed. Automatic anesthetic sprays spare them the disquietude which they would surely suffer on first acceleration and, after two long months, the flaming return to earth. All they have to do for this is to be good enough to permit their respiratory, cardiac, and other functions to be telemetered back to earth.

This reviewer is tired of seeing our simian brothers pushed around in the interests of scientific research. If this continues we may face the united hatred and despair of the monkeys of the world—a threat of no mean proportion when one considers their numbers and influence in the tropical world.

Mice could of course be used instead, since they are well-understood laboratory animals; their much lower weight (than the monkeys') would reduce the payload significantly and thus the requirements for the booster stages.

The article states succinctly that "in one move we'll crack the secrets of cosmic rays, meteor bullets—even lost isles," and "the monkeys will prepare the way for the men who follow." The vehicle, among many other functions, is supposed to count, by microphones, the number of micrometeorite hits on the vehicle's hull. This raises a neat problem, among others, of sorting out these signals from the inevitable noise prevalent on the telemetering channels.

**Comment:** A hair-raising technicolor journey guaranteed to frighten monkeys, impress the layman, and attract attention. Not liable to be launched by 1957.

*The James H. Wyld Project or The Model T Spaceship*—Proposed by the late J. Wyld, Reaction Motors, Inc.

Mr. Wyld about three years ago proposed in a number of talks a so-called "Model T" spaceship, partially of his own design and that of the R.M.I. Engineering Staff. The essential details of this project are:

<b>1st Stage:</b>			
(Thrust = 44,415 lb)	Propellants	14,805 lb	
	Structure	740	
	Power plant	529	
<b>2nd Stage:</b>			
(Thrust = 14,589 lb)	Propellants	4,863	
	Structure	243	
	Power plant	146	
<b>3rd Stage:</b>			
(Thrust = 5273 lb)	Propellants	1,758	
	Structure	88	
	Power plant	53	
		23,225	
	Payload	1,000	
	Total take-off weight	24,225	

The German V-2 weighed about 28,000 lb at take-off, so the rockets would be comparable in size, even after allowing some extra propellants in the spaceship to allow for maneuvering during descent.

Mr. Wyld stated that these figures would depend mostly on the attainment of extremely light structural weights, roughly  $\frac{1}{4}$  the corresponding figures for the V-2. He did not consider this unreasonable. This vehicle would cruise at an altitude of 200 miles and carry a payload of 1000 lb, including 2 lightweight (sic!) men of 100 lb each with food, water, and oxygen for two days, some supplies and instruments nose fairing, small retractable wings for gliding during descent, and a small insulated pressurized cabin.

The inclusion of two 100 lb men sounds unrealistic. A more reasonable plan would be to consider two women who are more likely to have weights of this order. This reviewer has not yet seen any practicing engineers even near this figure.

The weakest element in Wyld's proposal was the structural weight, or mass ratio factor (i.e., the ratio of the propellant weight to the total weight). A ratio of 0.8 is considered excellent for a rocket vehicle of this approximate size; the Martin Viking has an approximate ratio of this order and is considered an example of a well-designed unclassified high altitude test vehicle in this country. Even a five per cent improvement in this value would require unusually light materials and components and much extra design and development time. It does not seem at all likely that a ratio of 0.9 or more can be obtained without some revolutionary new approach to structural design.

*"Saloon," a Satellite Launched From a Balloon*, by Kurt R. Stehling, Bell Aircraft Corporation, and Raymond M. Misert, University of Iowa, *Aviation Age*, July 1955.

The authors discuss the possibilities and advantages of a high-altitude launched small satellite. It is shown that launching from a 3,000,000 cu ft balloon will permit a small three-stage vehicle of 13,500 lb to send 30 lb of instrumentation into an orbit. The first stage consists of four powder boosters weighing 12,000 lb with a thrust of 65,000 lb each, firing for 7 seconds.

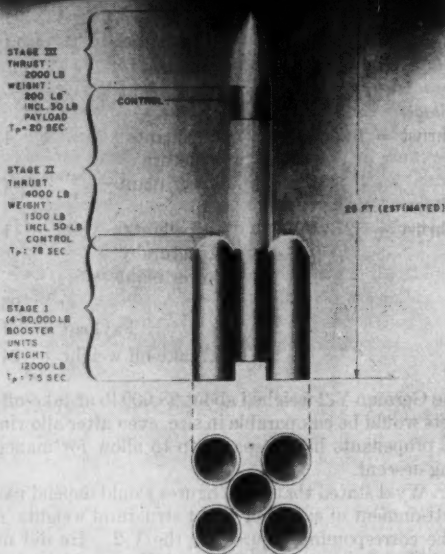
The second stage, containing 50 lb of controls, weight 1300 lbs., with a 4000 lb thrust liquid propellant engine firing for about 70 seconds. This stage reaches the actual orbit at a speed of approximately 17,000 ft/sec.

The third stage is the satellite, weighing 200 lb with 30 lb payload, continuing on in an elliptical orbit at 27,000 ft/sec. Based on unclassified information, this type of hardware might be available, although this is not certain.

The specifications for the powder boosters seem optimistic as is the statement that these might fire almost simultaneously. The problem of launching with a balloon would be very tricky, especially since the vehicle would have to fire in a



## PROPOSED SATELLITE VEHICLE

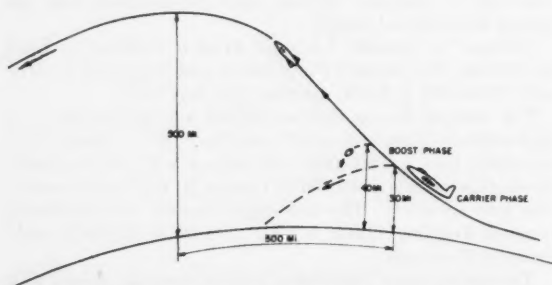


**Saloon.** An example of a possible arrangement of powder boosters and smaller second and third stage missiles which could be balloon-launched at a high altitude. Four 65,000, 7-sec boosters form the first stage.

noninhabited regional direction and preferably toward the east. Also, the launch angle of  $45^\circ$  would be critical.

**Comment:** A method of establishing a satellite if cheap hardware can be obtained and if everything works well. This proposal suggests launching the vehicle from a high altitude balloon in order to overcome the air drag which is suffered by rapid-acceleration powder booster vehicles. The balloon has about the same effect on the vehicle performance as a fourth stage would have, but without the high cost of such a stage. However, the problem of launching a huge, 3,000,000 cu ft balloon is very severe and would probably require several experimental flights.

*Some New Thoughts on Space Flight*, by P. E. Sandorff, M.I.T., ARS Preprint.



**Aircraft Launched Satellite.** The aircraft, traveling at a very high speed and altitude, is pulled up sharply to release the booster and satellite vehicle at the proper approach angle.

Professor Sandorff considers the possibility of launching a satellite vehicle with 500 lb payload from an aircraft (such as the B-52) from 50,000 ft. In this way, the aircraft, by lifting the vehicle through much of the atmosphere and adding its own speed, a satellite could be launched with a lower weight and cost than would be the case for a ground launched vehicle. He shows that a two-stage rocket, launched this way, weigh-

ing about 100,000 lb with 200 lb payload could achieve a 200-mile altitude orbit. In comparison, a three-stage rocket vehicle launched from the ground would weigh about 250,000 lb, with a corresponding increase in cost.

The aircraft would travel at 1100 mph and presumably carry the satellite in its bomb bay; at the critical launching moment, the aircraft would pull up sharply into a predetermined approach angle and then drop the satellite, which would continue under its own power.

Small rockets have, of course, been launched from aircraft for some time, as is evident from almost any news release on our interceptors. However, these have usually been relatively small rockets, launched from simple support brackets at fairly low speeds. Indeed, considering these techniques, it is possible to propose the firing of several small multistage satellite powder rockets, with a final payload of a pound or two, and reaching an orbit at 200 miles. A. J. Zaehring, Associate Editor of *JET PROPULSION*, has made several design studies of this type.

On the other hand, the launching of a giant rocket vehicle from a B-52, or any other huge high-speed aircraft, as stated by Sandorff, would require a very great development in launching and suspension techniques, especially at the high speeds mentioned in Sandorff's proposal.

## Thermodynamic Properties of Gas

(Continued from page 472)

tion to  $E - E_\infty$  at large  $v/v^*$  is small, and the error made is not as serious as one might conclude at first sight. In fact, by comparing the computed value using Equation [22] with the exact numerical value of the same function tabulated by Wentorf and collaborators, the difference is of the order of 10 per cent. Ten per cent accuracy is all we claim for the equation of state.

Similarly, we have the following formulas for molecular enthalpy  $H$ , molecular heat capacities,  $c_v$  and  $c_p$ , molecular entropy  $s$

$$(H - H_\infty)_T = 7(E - E_\infty)_T \dots \dots \dots [23]$$

$$(c_v - c_{v\infty})_T = \frac{k}{6} \frac{0.2314\eta - 0.177}{(0.278\eta - 0.177)^2} \dots \dots \dots [24]$$

$$(c_p - c_{p\infty})_T = 7(c_v - c_{v\infty})_T \dots \dots \dots [25]$$

$$(s - s_\infty)_T = \frac{k}{6} \frac{1}{0.278\eta - 0.177} - \frac{k}{0.177} \log \frac{0.278\eta}{0.278\eta - 0.177} \dots \dots [26]$$

The above formulas for heat capacities and entropy are less accurate than for energy and enthalpy because of the differentiation used in their derivations. But these equations are at least consistent among themselves.

## References

- 1 "Molecular Theory of Gases and Liquids," by J. O. Hirschfelder, C. F. Curtiss, and R. B. Bird, John Wiley, 1954, p. 263.
- 2 "An Equation of State Applicable to Gases at Densities Near That of the Solid and Temperatures far Above the Critical," by T. L. Cottrell and S. S. Paterson, *Proc. Roy. Soc. (A)* vol. 213, p. 214, 1952.
- 3 "High-Temperatures Equation of State by a Perturbation Method. I. Nonpolar Gases," by R. W. Zwanzig, *J. Chem. Phys.*, vol. 22, p. 1420, 1954.
- 4 "The Physics of High Pressures," by P. W. Bridgman, G. Bell and Sons, 1931, p. 108.
- 5 "Lennard-Jones and Devonshire Equation of State of Compressed Gases and Liquids," by R. H. Wentorf, R. J. Buhler, J. O. Hirschfelder, and C. F. Curtiss, *J. Chem. Phys.*, vol. 18, p. 1484, 1950.
- 6 "The Properties of Pure Liquids," by H. S. Tsien, *Journal of the American Rocket Society*, January-February 1953.
- 7 Ref. 1, p. 163.



## Gardner, Douglas, Kimball Speakers at Los Angeles Meeting



Trevor Gardner



Donald Douglas, Jr.



Dan A. Kimball

**T**HE Honorable Trevor Gardner, Assistant Secretary of the Air Force, will speak at the banquet to be held during the ARS 25th Anniversary Fall Meeting, Sept. 19-21, in Los Angeles at the Hotel Statler.

William J. Cecka of North American Aviation, Inc., chairman of the Meeting, has also announced that luncheon speakers will be Donald Douglas, Jr., vice-president, Military Relations, Douglas Aircraft, Inc.; and Dan A. Kimball, president, Aerojet-General Corp.

Five technical sessions will be held on Analysis, Combustion (two sessions), Field Problems, and Propellant Selection. Special trips are planned to the Walt Disney Studios, Aerojet-General Corp.,

the U. S. Naval Ordnance Test Station and Morris Dam.

The complete program is as follows:

### SUNDAY, SEPTEMBER 18

4:00 p.m. Registration

### MONDAY, SEPTEMBER 19

8:00 a.m. Registration

9:00 a.m. Session I

#### ANALYSIS

*Chairman:* Elmer P. Wheaton, Chief Project Engineer, Missiles, Douglas Aircraft, Inc.

*Vice-Chairman:* Edward Young, Douglas Aircraft, Inc.

Evaluation of Reliability in Product Development, M. Lipow and R. D. Geckler,

Aerojet-General Corp. (213-55)

Hydraulic Analogue for Transient Heat Transfer, E. L. Kumm and E. L. Knuth, Aerophysics Development Corp. (214-55)

Analysis of Thrust Chamber Performance Data, D. C. Schiavone, Bell Aircraft Corp. (215-55)

Generalized Solution for the Theoretical Thrust in a Rocket Motor for the C-H-N-O-F Atomic System, Mary Morgan, J. Silverman, and W. T. Weber. (216-55)

Generalized Theory of the Optimum Thrust Programming for the Level Flight of a Rocket-Powered Aircraft, P. Cicala, Politecnico di Torino, and A. Miele, Purdue University (217-55)

### 12:30 p.m. Luncheon

*Speaker:* Donald Douglas, Jr., Douglas Aircraft, Inc.

*Subject:* "The Role of the Airframe Company in the Development and Manufacture of Guided Missiles."

### 2:00 p.m. Session II

#### COMBUSTION

*Chairman:* William Bollay, president, Aerophysics Development Corp.

Application of Radiation Temperature Measurement Techniques to the Determination of Gas Temperatures in Liquid Propellant Flames, C. M. Auble and M. F. Heidmann, NACA, Lewis Flight Propulsion Lab. (218-55)

Experimental and Theoretical Aspects of Rocket System Stability, Y. C. Lee, Aerojet-General Corp. (219-55)

Solution of Thermo-Chemical Propellant Calculations on a High-Speed Digital Computer, A. J. Donegan, and M. Farber, Jet Propulsion Lab., CalTech (220-55)



## Satellites Interest Fort Wayne, Including "Mighty Mite"

Norman L. Baker, president of the Fort Wayne Section, appeared on television station WKJG-TV on July 14 and discussed unmanned satellites (above, left). On the same evening, Alfred J. Zaehring, president of the American Rocket Company and associate editor of JET PROPULSION, spoke to the Section on minimum satellites. Zaehring dealt specifically with his "Mighty Mite" (Minimum In Trajectory—Earth), a three-stage vehicle constructed of standard World War II solid propellant aircraft rockets, which can be launched from existing high alti-

tude aircraft.

Zaehring describes Mighty Mite as weighing less than a ton, having a payload of 5 lb of instruments. He estimates that 10 Mites would cost \$1 million and, due to their simplicity, that 50% of them could be expected to establish themselves successfully in an orbit.

Photo at right shows (left to right) Lloyd Wadekamper, vice-president of Section; Zaehring; Baker; Michael Katanik, treasurer; and Marion Stults, secretary.

# MISSILE SYSTEMS

## *Research and Development*

Broad interests and exceptional abilities are required of scientists participating in the technology of guided missiles. Physicists and engineers at Lockheed Missile Systems Division are pursuing advanced work in virtually every scientific field.

■ ■ ■

Below: Missile Systems scientists and engineers discuss future scientific exploration on an advanced systems concept with Vice President and General Manager Elwood R. Quesada. From left to right: Dr. Eric Durand, nuclear physicist, systems research laboratory; Ralph H. Miner (standing), staff division engineer; Dr. Montgomery H. Johnson, director, nuclear research laboratory; Elwood R. Quesada; Dr. Louis N. Ridenour (standing), director, program development; Willis M. Hawkins (standing), chief engineer; Dr. Joseph V. Charyk (standing), director, physics and chemistry research laboratory; Dr. Ernst H. Krause, director, research laboratories.

■ ■ ■

Scientific advances are creating new areas of interest for those capable of significant contribution to the technology of guided missiles.

*Lockheed* **MISSILE SYSTEMS DIVISION**

*research and engineering staff*

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# AMERICAN ROCKET SOCIETY

## FALL MEETING

Los Angeles

September 19-21

The technology of guided missiles is literally a new domain. No field of science today offers greater scope for creative achievement.

Physicists and engineers with special abilities applicable to this field will be interested in new developments at Lockheed Missile Systems Division.

A.A. Daush, R.B. Lang and senior members of the technical staff will be available for consultation at the convention hotel. For interview phone MADison 9-3863.

*Lockheed*

MISSILE SYSTEMS DIVISION

Experiments on Burning of Single Drops of Fuel in Air at Various Ambient Conditions, M. Goldsmith, Rand Corp. (221-55)

7:00 p.m. Tour of Walt Disney Studios  
Buses to leave from Francisco Street entrance.

### TUESDAY, SEPTEMBER 20

8:00 a.m. Registration  
9:00 a.m. Session III

#### COMBUSTION

Chairman: Norman Reuel, Propulsion Field Lab., North American Aviation, Inc.

Determination of the Effective Specific Heat and True Gas Composition in a Rocket Motor, P. J. Blatz, Aerojet-General Corp. (222-55)

The Effect of a Variable Evaporation Rate on the Ballistics of Droplets, C. C. Miesse, Aerojet-General Corp. (223-55)

Interference During Droplet Burning, J. F. Rex, A. Fuhs, and S. S. Penner, Cal-Tech (224-55)

Two Methods for Measuring Ignition Delay of Self-Igniting Rocket Propellant Combinations, D. J. Ladanyi, NACA, Lewis Flight Propulsion Lab. (225-55)

Testing 100,000 Horsepower Jet Engines, L. E. Dunn, Marquardt Aircraft Co. (226-55)

12:30 p.m. Luncheon

Speaker: Dan A. Kimball, Aerojet-General Corp.

2:00 p.m. Session IV

#### GENERAL PROBLEMS

Chairman: Robert Terbeck, Jet Propulsion Lab., CalTech.

Missile Flight Safety Considerations at White Sands Proving Ground, G. L. Meredith and D. I. Thompson, Operations Branch, Hq., White Sands Proving Ground (227-55)

Operational Analysis of a Field Support Problem for a Complex Weapon System, A. Shapero and J. Dresner, Hughes Aircraft Co. (228-55)

The Problem of Rocket Engine Component Testing, F. A. Jennings, North American Aviation, Inc. (229-55)

Perturbation Analysis of Low Frequency Rocket Engine System Dynamics on an Analog Computer, B. N. Smith, North American Aviation, Inc. (230-55)

Missile Range Instrumentation, J. P. Judin,

Head, Instrument Operations Division, U. S. Naval Ordnance Test Sta. (231-55)

6:00 p.m. Reception and Dinner

Toastmaster: Richard W. Porter, President, ARS

Speaker: The Hon. Trevor Gardner, Assistant Secretary of the Air Force

Subject: "Guided Missiles: Reliability and Associated Problems."

### WEDNESDAY, SEPTEMBER 21

8:00 a.m. Registration  
9:00 a.m. Session V

#### PROPELLANT SELECTION

Chairman: Howard Seifert, The Ramo-Wooldridge Corp.

The Use of Weapon Systems Engineering Concepts in the Design of Solid Propellant Rocket Power Plants, H. L. Thackwell, Grand Central Rocket Co. (232-55)

A New Economic Approach to Rocket Propellant Selection, H. R. Biederman and H. M. Kindsvater, Lockheed Aircraft Corp. Missile Systems Div. (233-55)

Some New Metallurgical Processes of Interest in the Field of High Speed Flight, E. C. Bishop, Westinghouse Electric Corp. (234-55)

The Manned Satellite—A Stepping Stone to Space Flight, K. A. Ehricke, Convair (235-55)

Afternoon—Trips to Aerojet-General Corporation, Morris Dam, and Naval Ordnance Test Station

7:30 p.m. Movies

## Thirteen Sessions Slated for Chicago

THE 25th Anniversary Annual Convention of ARS, to be held in conjunction with the Diamond Jubilee Annual Meeting of ASME, will take place November 14-17 at the Hotel Hilton in Chicago.

Kenneth H. Jacobs of Chicago Midway Laboratories, chairman of the Meeting, announces that 13 sessions are scheduled. They will include Thermodynamics and Heat Transfer, Rocket System Stability, Combustion, Space Medicine, Solid Propellants, Aerodynamics, Liquid Propel-

### SECTION PRESIDENTS

Alabama: JOSEPH WIGGINS, Thiokol Chem. Corp.; Arizona: CHAS. J. GREEN, Hughes Aircraft Co.; Central Texas: B. S. ADELMAN, Phillips Petroleum Co.; Chicago: V. J. CUSHING, Armour Research Foundation; Cleveland-Akron: W. T. OLSON, NACA; Detroit: FRED KLEMACH, Nal, Research & Development Corp.; Florida: R. S. MITCHELL, Pan American World Airways; Fort Wayne: NORMAN BAKER, Indiana Technical College; Indiana: P. M. DIAMOND, Purdue Univ.; Maryland: W. G. PURDY, Glenn L. Martin Co.; National Capital: E. C. PAGE, Page Communications, Inc.; New England: C. LINCOLN JEWETT, Arthur D. Little, Inc.; New Mexico-West Texas: R. C. SHERBURNE, New Mexico A & M; New York: C. W. CHILLSON, Curtiss-Wright Corp.; Niagara Frontier: T. ZANNES, Bell Aircraft Corp.; Northeastern New York: A. H. FOX, Union College; Northern California: M. A. PINO, California Research Corp.; Pacific Northwest: R. M. BRIDGFORTH, Boeing Airplane Co.; Princeton Group: IRVIN GLASSMAN, Princeton Univ.; St. Louis: NORTON B. MOORE, McDonnell Aircraft Corp.; Southern California: C. M. McCLOSKEY, ONR; Southern Ohio: W. J. MIZEN, Bendix Aviation Corp.; Twin Cities: J. J. SCHONS, Univ. of Minnesota.

### ARS MEETINGS CALENDAR

Sept. 19-21 ARS Fall Meeting, Los Angeles, Calif.  
Nov. 13-18 ARS-ASME Annual Convention, Chicago, Ill.  
Dec. 12-16 Nuclear Science and Engineering Congress, Cleveland, Ohio

Abstracts should be submitted to the Program Chairman, American Rocket Society, 500 Fifth Ave., New York 36, N. Y., six months prior to the meeting date.





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In addition, a special forum on standardization of rocket symbols will be offered, sponsored by the American Standards Association Subcommittee on Letter Symbols for Rocket Propulsion (ASA Y10-17).

The complete program will be carried in the November issue of *JET PROPULSION*.

### How WSSCA Works at White Sands

C. E. DUCKETT, of White Sands Signal Corps Agency was the speaker at a meeting of the New Mexico-West Texas Section on July 28 at New Mexico A. & M. A. Subject of the talk was "The Signal Corps and Guided Missiles."

He pointed out that WSSCA is responsible for a wire communications network on the proving grounds range which compares in complexity with a telephone system serving several small cities, including long-distance exchanges. In addition, a 23-channel microwave relay network is maintained, transmitting information from five locations. This network is equivalent to a system of over 100 television stations.

Frequency interference at WSPG becomes a major problem. Consequently, WSSCA constantly monitors all important frequencies, establishes compromise agreements regarding frequency usage with all radiating locations in the area, schedules frequency usage on the proving ground.

A chain radar tracking system is maintained by WSSCA to provide continuous tracking of missiles. A range safety device destroys missiles if they go outside the proving ground.

WSSCA provides information on prevailing surface winds for missiles which are not guided and is involved in a program of upper atmosphere research and an electronics warfare program.

The Signal Corps Engineering Laboratory performs tests on communications equipment under extreme climatic conditions, and will participate in the International Geophysical Year (1957-1958) at Fort Churchill, Canada.

Motion pictures of Aerobee rocket firings and of equipment tests at Fort Churchill, Canada, illustrated the talk.

A June 30 meeting of the Section featured Herbert L. Karsch, assistant to the Commanding General at WSPG, who spoke on the Henschel series of guided missiles.

These World War II German missiles were developed for air-to-air, ground-to-air, air-to-ground and air-to-water use. Slides illustrated the talk.

### Stapp Leader of Holloman Group

THIS year's officers of the Holloman Group of the New Mexico-West Texas Section include Col. John P. Stapp, renowned Chief of the Aero Medical Field Lab at Holloman Air Development Center, as president; Walter L. Andre, vice-president; and J. A. McCurdy, secretary treasurer.



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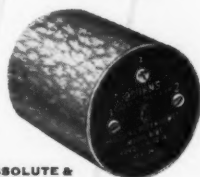
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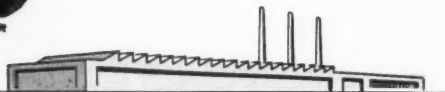
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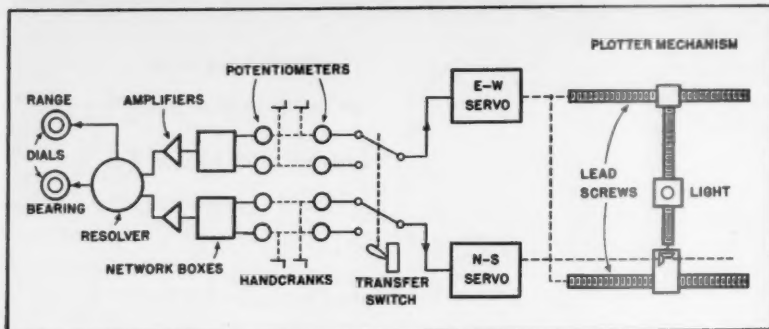
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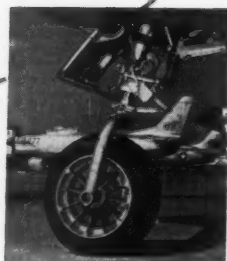
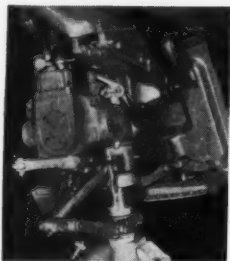
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## Book Reviews

C. F. Warner, Purdue University, Associate Editor

**Introduction to Astronomy**, by Cecelia Payne-Gaposhkin, Prentice-Hall, Inc., New York, 1954, 481 pp. \$8.

Reviewed by C. L. BROWN  
 Purdue University

"Introduction to Astronomy" is, according to the author, intended "to introduce the elements of astronomy to the student and the general reader who may have little background in mathematics or physics." My only criticism of the book is on this point. I consider it the finest first text in astronomy that I have encountered, but I fear that a reader whose physics and mathematics are of high-school level will be bewildered at frequent intervals. Not that Dr. Payne-Gaposhkin has not tried valiantly to explain these matters in simple terms; she simply refuses to compromise with facts and resort to half truths. The book is excellent for a reader with a background of basic calculus and college physics.

The text is divided into three sections: introduction, the solar system, and stellar astronomy. The discussion of the earth from the point of view of an astronomer is quite interesting and includes an unusually complete discussion of the various motions of the earth. A chapter on "Tools and Methods" includes a precise and lucid exposition of celestial surveying and a short but detailed section on astronomical instruments and their use. The mysteries of right ascension and declination are clearly outlined. Telescopes, spectroscopes, and other instruments are described with an eye to an understanding of the principles by which they operate.

Our solar system receives less attention than is usual but a considerable amount of recent data is given. The remainder of the book is devoted to stellar astronomy. Astrophysics receives considerable attention, and a good account of star interiors, including energy production reactions, is presented. The variable stars, the astronomical yardsticks, are examined much more carefully than is usual in a book at this level.

An engineer with little knowledge of astronomy should find "Introduction to Astronomy" a valuable addition to his library if he wishes to begin a study of the subject. Without intending disrespect to other members of the profession, I would call the author a "working" astronomer. By this I mean that her primary interest is in the accumulation of more and more facts about the universe by new instruments and methods and the organization of these facts into a coherent pattern. Her writing reflects this interest.

**Thermodynamics**, By R. L. Sweigert and M. J. Goglia, The Ronald Press Co., New York, 1955, 355 pp. \$6.50.

Reviewed by C. F. WARNER  
 Purdue University

This textbook was written for a first course in engineering thermodynamics for

engineering students having completed courses in college physics, calculus, and preferably one course in differential equations. While some other textbooks in this field claim to make some use of calculus and differential equations, the subject matter of thermodynamics is based upon a mathematical presentation in this book. The mathematical development of the many equations is presented in sufficient detail so that the engineering student may follow the logical steps with ease. The numerical examples aid the reader in acquiring competence in the solution of common engineering thermodynamic problems.

Since use is made of differential equations in the very beginning, the Maxwell relations are used extensively in the treatment of properties of pure substances and the perfect gas. It is the opinion of this reviewer that this type of presentation aids the student greatly in acquiring an understanding of the interrelationships between the physical properties of pure substances and the equations available for obtaining approximate values of properties in the absence of reliable data. The numerous three-dimensional  $p$ - $v$ - $t$  surfaces of various substances are superior to the conventional two-dimensional drawings in explaining the behavior of engineering fluids as the pressure, temperature, or volume is varied.

The presentation of the second law of thermodynamics and the subjects of entropy and reversibility are quite similar to the treatment of the same subjects by Keenan. The use of the dashed line to signify an irreversible process is most helpful in differentiating between reversible and irreversible processes. The chapter on Gas Dynamics is limited to the consideration of one-dimensional steady state flow, and the material presented is similar to that used by both N. A. Hall and A. H. Shapiro in their respective treatments of the same subject matter.

The chapter on Thermodynamics and Chemistry includes such topics as equilibrium constants and the maximum adiabatic flame temperature. The material presented is in sufficient detail so that the student will be able to understand the simpler combustion reactions involving carbon, hydrogen, and oxygen atoms.

The majority of the engineering applications are presented in the final chapter entitled Applied Thermodynamics. This organization of the subject material enables the authors to divorce the basic topics from specific pieces of engineering equipment. Some will criticize the authors for the brevity of this chapter. Throughout the text excellent illustrative examples are given and twenty or more problems are included at the end of each chapter. Many of these problems illustrate points not completely developed in the text material so that the reader is encouraged to advance upon his own.

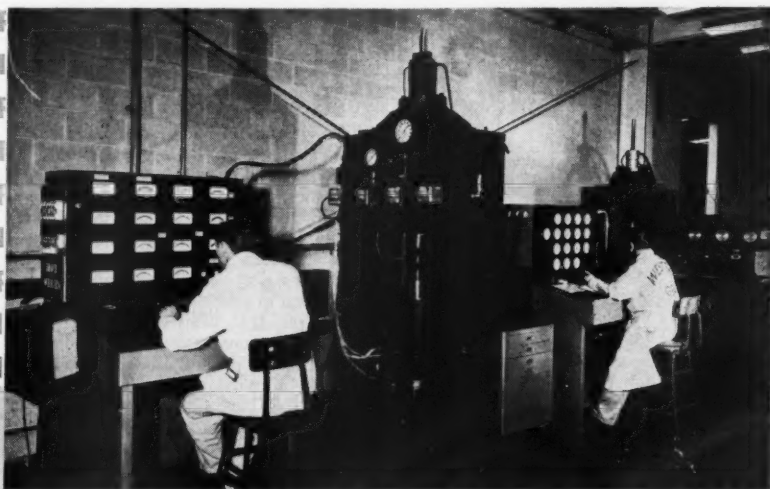
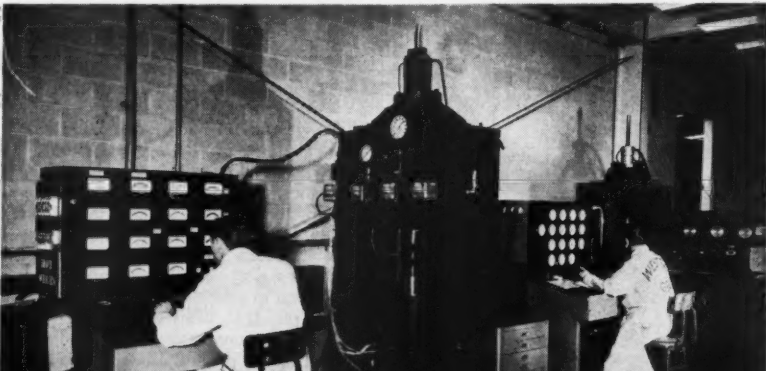
It should not be construed that this is an advanced text or reference book, for it



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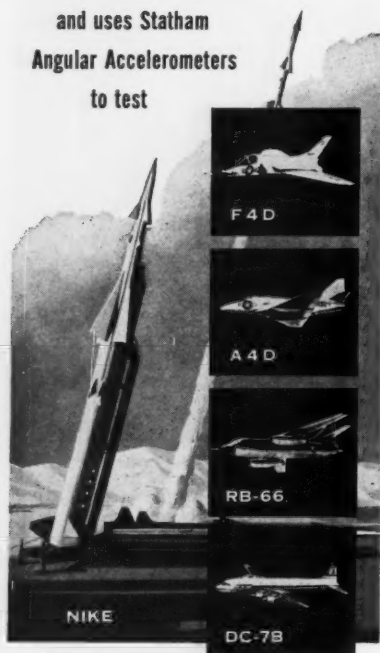
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is not; however, it is an excellent basic treatment of thermodynamics for both the engineering student and for the older engineer wishing to review the subject. This reviewer wishes to point out that while he approved of the mathematical presentation, in his opinion more distinction should be made between the mathematical operator  $d$  or differential and a small quantity of heat and work, since these latter two quantities are not exact differentials.

**Energy Transfer in Hot Gases**, National Bureau of Standards Circular 523, United States Government Printing Office, 1954, 126 pp. \$1.50.

Reviewed by H. F. CALCOTE  
Experiment Incorporated

This book is a compilation of sixteen papers presented in 1951 on the occasion of the 50th anniversary of the establishment of the National Bureau of Standards. The Symposium on Energy Transfer in Hot Gases was the sixth in a series of twelve conferences held to celebrate the event. Although this little book is late in arriving after the conference, it is nevertheless helpful in that it brings together the subject matter as of that date.

Most of the papers concern themselves with spectroscopy and combustion flames and phenomena arising therefrom. In addition there are two papers on the theory of flames, a paper on the temperature of the hydrogen fluorine flame, and a final paper on the astrophysicist's concept of temperature.

Gaydon discusses processes of electronic excitation in relation to flame spectra and points out that most optical methods of temperature measurement depend on electronic excitation and that if equilibrium is not established, this is likely to be closer to the vibrational than to the translational temperature of the gas. Hornbeck and Herman present some experimental studies on polyatomic flame bands and point out the importance of using low-temperature high intensity sources in studying polyatomic flame spectra. Ignition spectra of polyatomic free radicals are briefly discussed by Dyne. The distribution of OH rotational intensities in flames obtained with a high-resolution grating monochromator are employed by Broida to measure the rotational temperatures and as a means of investigating thermal equilibrium in hot gases. Penner, Gilbert, and Weber in spectroscopic studies of low-pressure flames of propane oxygen find abnormal rotational excitation and temperatures of 5500 to 8000°K for the upper rotational states of OH. In a paper on "Studies of Emission and Absorption in Flames," Dieke and Crosswhite discuss the difficulties involved in determining energy distribution in flames. The energy distribution of CO molecules in carbon monoxide-oxygen flame is discussed with respect to equilibrium and nonequilibrium phenomena by Silverman. Benedict and Pyler summarize recent work in high-resolution spectra of hydrocarbon flames in the infrared and find that in every case where a "temperature" could be measured from resolved spectra, the results were from 2400 to 2800°K, whereas visible and ultraviolet spectra from the same regions

of comparable flames yielded values greater than 3000°K. The lower-lying excited states responsible for infrared emission show a much closer approach to complete thermodynamic equilibrium than states with higher amounts of electronic excitation. Penner outlines the basic theoretical relations which are used to calculate the infrared emissivity of diatomic gases. The status of work on a program aimed at measuring the infrared absorption spectra of gases heated under controlled conditions of temperature, pressure, and composition is described by Tourin. Reactions of atomic hydrogen in flames, particularly chemiluminescent phenomena are discussed by Arthur and Townend. Wilson, Conway, Englebrecht, and Grosse calculate the maximum theoretical flame temperature for hydrogen fluorine to be 4300°K and, employing the sun as a comparison source with the line-reversal method, find an experimental value of  $4300^\circ\text{K} \pm 150^\circ$ . The paper by Hari K. Sen on the astrophysicist's concept of temperature emphasizes the care which one must take in employing the word "temperature" in nonthermodynamic case. This is a very appropriate consideration on which to end this interesting booklet.

**Servomechanisms**, by John C. West, English Universities Press Ltd., London, England, distributed by the MacMillan Company, New York, 1954, 238 pp. \$5.

Reviewed by E. M. SABBAGH  
Purdue University

Control engineering was an art to be gained by experience only. Depending upon its use, it had acquired different terminology in different industries. Not until the early 1930's had this art become a science of extreme importance because of its application for military and industrial use and is now undergoing a unification of terminology and symbols.

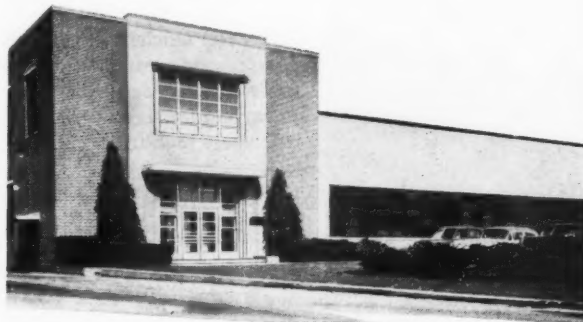
As a science, control engineering uses the basic theories developed by the electrical power and communication engineers. Because of its importance, electrical engineering schools the world over are now offering courses in control and servomechanisms. Control engineering, however, is not confined to the electrical field. It comprises all branches of engineering and uses all kinds of equipment.

This volume on servomechanisms is an elementary textbook intended for undergraduate students interested in control. These students should have a knowledge of electronics, the equivalent of a course in mathematics covering calculus through ordinary differential equations, and some circuit theory. The absence in the content of the text of any description of hydraulic, pneumatic, or other non-electrical components leads one to believe that the book is written for electrical engineering students only. The book, however, may prove useful to all engineers who want to have a knowledge of the basic theory of servomechanisms and control.

In an early chapter the author defines servomechanisms and proceeds to explain by illustration a simple position control system. He then studies damping and shows its effect on the operation of the

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system. He explains how damping can be gotten. Hurwitz-Routh stability conditions and the Nyquist diagram are then introduced. Bode theorems relating attenuation and phase angle for minimum phase networks are explained.

Throughout the text the author applies the theories introduced to his model.

In later chapters the author discusses various electrical error detectors, some electronic amplifiers, and some rotary rectifiers of the single and two-stage type.

In chapter 19, the author shows how, through conformal mapping or other operations, the transient response of a system could be obtained from the open loop frequency response.

Throughout the book and where convenient the author makes use of the phase plane to get a better understanding of the principles involved.

**Optical Instrumentation**, Edited by George S. Monk and W. H. McCorkle, McGraw-Hill Book Co., Inc., New York, 1954, 262 pp. \$3.75.

Reviewed by I. WALTERSTEIN  
Purdue University

This is the first volume in the National Nuclear Energy Series aimed at describing those phases of the research and development of the wartime work in the field of nuclear energy which have since become declassified. About one third of the book is devoted to a survey of the optical problems which arose in connection with the need to view various irradiated areas separated from the observer by heavy extended shields. The remainder consists of reports, giving details of design and performance of various instruments and optical materials employed to surmount the obstacles that arose.

Some of the problems which were investigated in connection with fluorescence, preparation of reflection surfaces, coloration of optical materials, etc., may not be of primary interest to readers of this journal. The sections on periscopic instruments, wide angle viewers, and borescopes are of much more general interest. Specifically the instruments described are designed either to view close-ups or scan a wide angular field. By the repeated use of optical extender sections and reflectors some of the devices can be made of almost arbitrary length or shape. Difficulties of procurement during the war were overcome by versatile utilization of available optical elements originally designed for entirely different requirements. The basic aim is that of viewing an inaccessible area. Here the lack of access was due to radiation hazard, but whatever the cause may be, several basic guides for design are given which can be modified for other needs (exclusive of those where explosive and shock effects are dominant). While most of the designs are for visual observation, several are shown modified for photographic recording.

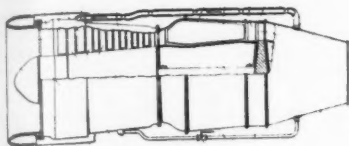
### Book Notices

A booklet containing nine papers presented at the Thermal Barrier Symposium of the ASME held in New York in December 1954 may be obtained from the Office of Information Services of New York University.



# New Patents

George F. McLaughlin, Contributor



2,712,727

Starting equipment for resonant pulse jet combustion device (2,710,524). Paul A. Frank and Walter L. Rhodes, Troy, Ohio, assignors to Wm. L. Tenny.

Electrical generating means, connected to an igniter to initiate combustion, and driven in direct response to manual operation of an air pump.

Gas turbine power plant (2,711,631). Andrew V. D. Willgoos, deceased, late of W. Hartford, Conn., assignor to United Aircraft Corp.

Gas turbine power plants with means for preventing or removing ice formation (2,712,727). Frederick Wm. W. Morley and Ernest Freeland-Upshall, Clitheroe, England, assignors to Rolls-Royce, Ltd.

Conduits supplying compressed air passed in heat exchange relation with the exhaust gas delivered to a chamber to heat the wall structures having external surfaces requiring anti-icing during operation.

Gas turbine apparatus (2,712,218). John A. Ritter, Norwood, Pa., assignor to Westinghouse Electric Corp.

Scavenging system including a fuel system valve for diverting unburned fuel from the combustion zone when the fuel flow stops, and ejection means for facilitating withdrawal of fuel by way of the valve.

Liquid fuel control means for aerial bodies of the ram-jet type (2,712,219). Eugene Harold Warne, Birmingham, England, assignor to Joseph Lucas Industries, Ltd.

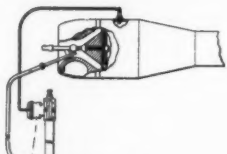
Servomechanism operated delayed action throttle in series with a primary throttle which is opened by a valve actuated by gas pressure and a difference in fuel pressure.

Arming mechanism (2,712,806). Arthur V. Hughes, Sharon, Pa., assignor to the U. S. Navy.

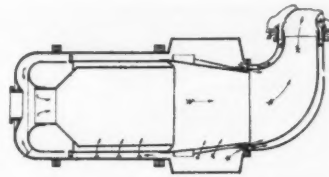
Control system for a jet propelled hydro-bomb, with means for removal of a low resistance shunt, connected with the igniter, when the bomb reaches a given distance from the launching station.

Rocket and turbine engine combination for aircraft (2,713,243). Elliott O. Seaver, Caldwell, N. J., assignor to Curtiss-Wright Corp.

Rocket motor, within the tail cone of a gas turbine engine, having selectively operable liquid fuel and liquid oxidizer feed mechanisms driven by the turbine.



2,710,524



2,712,729

Cooling systems of gas turbines (2,712,729). Raymond E. Wigg, Lincoln, England, assignor to Ruston & Hornsby, Ltd.

An intermediate pressure-resistant casing surrounding the thin-walled duct of a combustion chamber and separated from the duct by two spaces, the first of which is in communication with the compressor outlet but having itself no outlet, so that internal pressure in the first space approximates that of gas within the duct.

Method of operating a jet engine (2,712,726). Homer M. Fox, Bartlesville, Okla., assignor to Phillips Petroleum Co.

Incorporation into liquid hydrocarbon fuel a viscosity index improver selected from the group consisting of high molecular weight polymers, one having a Saybolt viscosity of 100 degrees F of the order of 30,000 and at 210 degrees F of the order of 1900.

Ammunition retainer for rocket launchers (2,712,270). Samuel G. Green, Gray, Ga.

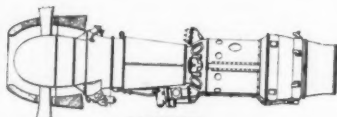
Contact on a retainer detachably secured across the breech of a launcher for fin-stabilized rocket propelled missiles. Means for detachably engaging certain of the fins to secure the missile in firing position. Safety mechanism has a movable contact biased to engage the retainer contact in the firing position, and connects with a power source for energizing the igniter.

Gas turbine afterburner apparatus (2,712,221). Walter D. Pouchot, Prospect Park, Pa., assignor to Westinghouse Electric Corp.

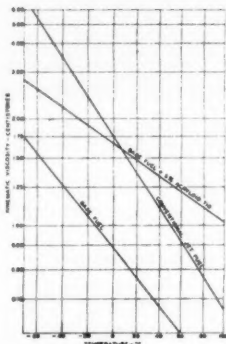
Fuel vaporizing units in a combustion chamber including hollow fairings having upstream openings for receiving gases, a backplate closing each fairing adjacent to the downstream outlets, fuel intakes on each fairing, and evaporating surfaces between the intakes and outlets.

Propellant assembly for jet propelled device (2,712,283). Sidney Golden, Cumberland, Md., assignor to the U. S. Navy.

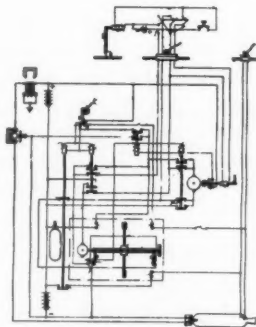
Reaction motor propellant charge consisting of perforated disks of powder loosely retained on a rod, the outer curved areas of the disks having means to prevent their combustion.



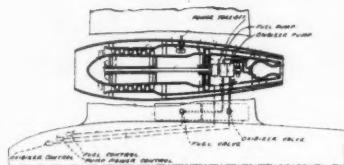
2,711,631



2,712,726



2,712,806

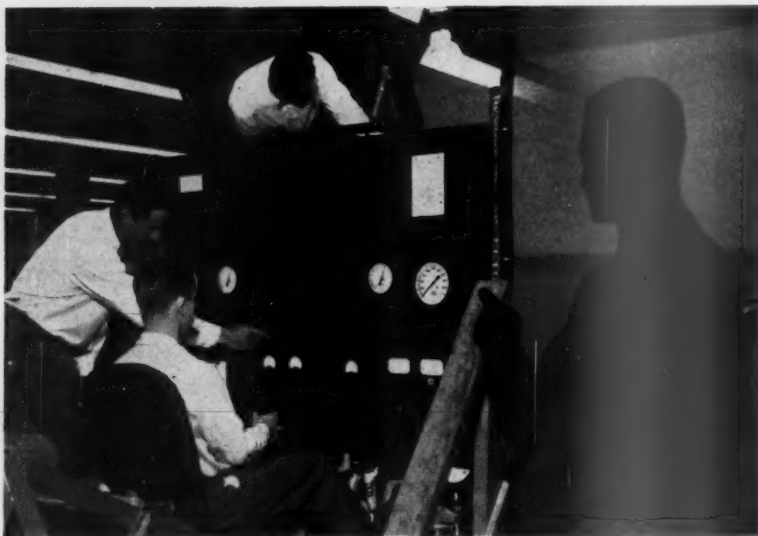


2,713,243

(Continued on next page)

EDITOR'S NOTE: The patents listed above were selected from recent issues of the Official Gazette of the U. S. Patent Office. Printed copies of patents may be obtained at a cost of 25 cents each, from the Commissioner of Patents, Washington 25, D. C.

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### New Patents (Cont.)

**Mechanism for reducing the drag of ram-jet engines (2,712,217).** Elbert R. Sargent and Edwin P. Neikirk, Ferguson, Mo., assignors to McDonnell Aircraft Corp.

Piston-actuated inlet doors normally subject to air pressure, held open while fuel is being supplied under pressure to the nozzle, and actuated to closed position when the flow of fuel is stopped.

**Jet fuel (2,712,496).** Sol Skolnik and Howard W. Kruse, Inyokern, Calif., assignor to the U. S. Navy.

Fuel comprising approximately 82 per cent aniline by weight and 18 per cent hydrazine by weight.

**Jet engine fuel (2,712,497).** Homer M. Fox and Sylvester C. Britton, Bartlesville, Okla., assignors to Phillips Petroleum Co.

Fuel having a boiling point between 90 degrees F and 500 degrees F, consisting essentially of hydrocarbon fuel materials and at least one aromatic hydrocarbon gum inhibitor.

**Gas turbine inter-stage reheating system (2,712,728).** Warren K. Lewis and Edwin R. Gilliland, Arlington, Mass., assignors to Esso Research and Engineering Co.

Means for returning partially expanded gas from one of the turbine stages to a surface heater, and of returning the gas to a succeeding stage. Also, means for conveying gaseous combustion products from the combustion chamber to the turbine.

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# Technical Literature Digest

M. H. Smith, Associate Editor, and M. H. Fisher, Contributor  
The James Forrestal Research Center, Princeton University

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